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REPORT NO. CASD-NSC73-005
CONTRACT N62269-72-C-0414

FBR 5858

DESIGN AND FABRICATION OF A BORON ALUMINUM COMPOSITE WING BOX TEST SPECIMEN

FINAL REPORT

July 73

GENERAL DYNAMICS
Convair Aerospace Division

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Prepared Under
Contract N62269-72-C-0414

Prepared for
NAVAL AIR DEVELOPMENT CENTER
Air Vehicle Technology Department
Warminster, Pennsylvania 18974

Approved for Public Release, Distribution Unlimited

Prepared by
GENERAL DYNAMICS CONVAIR AEROSPACE DIVISION
San Diego, California

FOREWORD

This report was prepared by Convair Aerospace Division of General Dynamics, Advanced Composites group, San Diego, California, under the terms of Contract N62269-72-C-0414.

This final report covers the entire program from March 1972 through May 1973. The program was sponsored by the Air Vehicle Technology Department, Naval Air Development Center (NADC) Warminster, PA 18974. Mr. Anthony Manno, Code 3033 was the Project Engineer for NADC.

The following Convair Aerospace personnel were the principal contributors to the program: G. F. Foelsch, Stress Analysis; R. C. Savary, Design; C. R. Maikish, Composite Subcomponent and Component Fabrication; C. L. Bennett, Steel Fabrication, and W. M. Parker, Subcomponent Testing; Messrs. W. Kowel and D. R. Turner of Canadair Limited also contributed to the design and analysis of the wing box. Mr. W. F. Wennhold was the program manager for Convair.

The static and fatigue testing of the wing box was conducted at NADC under the direction of Mr. Anthony Manno.

All the photographs in Section 5 of this report are official U. S. Navy photographs by Mr. A. Shanks.

SUMMARY

This report summarizes the design, analysis, fabrication and testing of a wing box specimen utilizing boron aluminum as the primary structural material.

The primary purpose of the program was to demonstrate the feasibility of utilizing boron-aluminum in a major aircraft structural component. The wing section selected is typical of many aircraft wing structures and simulated an integral wing fuel tank section capable of withstanding internal pressure in addition to the basic bending and torsion loads.

The wing section was not intended to typify any specific aircraft nor was it intended to solve any design problem associated with any specific aircraft. Rather, it was to demonstrate, as the next logical development step, the ability of boron-aluminum to be used in a lightweight primary aircraft structure.

The design of the wing section was based upon the design requirements for the graphite epoxy wing box designed and built by Convair Aerospace in 1969 and tested by the NADC early in 1970.

A detailed stress analysis of the test specimen was performed. Computer programs were utilized to generate internal member loads and web shears, and to analyze load introduction areas. Conventional aircraft stress analysis techniques were utilized throughout.

Subelement specimens of the stringers and load introduction areas were fabricated and tested to verify the integrity of the concept prior to fabrication of the actual test specimen. The successful testing of these specimens demonstrated the adequacy of the design approach.

The wing box test specimen was fabricated using the latest metal matrix composite fabrication techniques which, for the most part, are based upon conventional sheet metal technology.

The specimen was structurally tested at the Naval Air Development Center, Warminster, Pennsylvania, where it successfully withstood 150% of limit loads in bending, torsion, combined bending and torsion, and internal pressure.

The specimen was then subjected to a Navy fighter fatigue spectrum for over five 6,000 hour life times. Failure occurred at approximately 31,000 equivalent flight hours and consisted of longitudinal splitting of several stringers and the skin along rivet rows. This allowed the compression skin and stringers to buckle locally. The nature of the failure was such that the tension capability of the compression skin was not impared and a reverse bending load case test was performed.

During the subsequent reverse bending testing the wing failed at 93% of the predicted failing load with failure consisting of buckling of the skin and stringers on the entire compression side.

The program demonstrated that a structurally sound wing section utilizing boron/aluminum as the primary material could be fabricated utilizing available sheet metal technology. The completed wing with a 25% weight saving over a conventional aluminum wing met all static load requirements and demonstrated a fatigue life far in excess of expected future requirements.

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INTRODUCTION

Boron/aluminum has had considerable development in the last several years, but needs ultimately to be applied to actual flight hardware to prove it possesses the reliability to provide high-strength light weight structure for future applications. It combines high specific mechanical properties with superior fatigue qualities, environmental resistance and compatiblity with existing structural concepts.

The work reported here is a natural extension of the work previously accomplished on other composite programs at Convair Aerospace and was considered to be one of the next logical steps toward the ultimate goal of flight qualified boron/aluminum primary structure.

The design of the wing box was based upon and used the same basic geometry as the graphite-epoxy wing box built by Convair Aerospace under IRAD funds and tested by NADC. The same design criteria was used for the B/Al wing box except that an internal pressure loading condition and a negative bending moment condition was used to make the design conditions more representative of an actual aircraft structure. The analytical work done for the graphite wing served as a basis for the boron/aluminum wing box analysis and enabled the structural analysis to be performed expeditiously.

Several approaches to the methods of fabricating metal matrix composite structures have been taken by the aerospace industry. The methods which appear to have the greatest potential of producing structures at costs comparable with conventional metals are those which are based upon conventional sheet metal technology.

The wing box was therefore planned and designed to take advantage of this and to further demonstrate that a relatively complicated structure such as a wing box could be fabricated from boron/aluminum by these methods. No new methods were planned nor needed. All manufacturing methods used were merely extensions of those presently in use in any aerospace factory.

The philosophy used in the design of the wing box was to provide a dependable structure within a limited budget. Accordingly conservative assumptions and practices were used throughout the entire program. The initial estimates for stringer gages resulted in stiffeners which were stronger than necessary. There was not sufficient time nor budget to permit redesign nor rerunning of the subelement tests after the initial tests reported here. This practice was true during all the design and analysis. The design was changed only when the analysis showed a negative margin. This practice is defended in that it is similar to that applied to an actual aircraft.

DESIGN AND ANALYSIS

2.1 DESIGN PHILOSOPHY

The wing box was intended to demonstrate the feasibility of utilizing boron aluminum in a typical wing structure. Accordingly, this material was utilized to the maximum extent possible where it would demonstrate an advantage over conventional metals.

Other materials, aluminum, titanium, corrosion resistant steel, and steel were utilized for fillers, clips, brackets and load introduction members.

A conservative approach was used in the design of the test specimen when ever possible. The skins were designed to buckle above limit load both in bending and torsion. The spars, however, were designed as sandwich panels to minimize problems associated with designing a pressure resistant flat spar.

The allowable stresses used for the boron aluminum material are considered to be very conservative. This was primarily because of the lack of sufficient data on boron aluminum under combined states of strain.

2.2 DESIGN CRITERIA

The wing box which is the subject of this report was intended to demonstrate the feasibility of using metal matrix composites for a typical aircraft wing structure. The design criteria used here is based upon those used for a similar graphite/epoxy wing box built by Convair Aerospace and tested by NADC (Ref. 1). The loads for that structure were derived from the loading conditions on the outboard wing of the S3A antisubmarine aircraft. This original criteria was somewhat modified and updated by the addition of a negative bending moment and internal pressure loads. The latter requirement was to simulate a fuel carrying wing.

2.2.1 LOADS.

Separate

540,000 in.-lb. ult. positive bending moment

270,000 in.-lb. ult. negative bending moment

360,000 in.-lb. ult. torsion moment

10 psi ult. pressure

Combined

540,000 ult. bending with 50,000 ult. torsion
and 10 psi internal pressure.

2.2.2 REQUIREMENTS:

a. The design to simulate a typical subsonic aircraft wing section with integral fuel tanks.

b. An access door to be provided in the lower surface.

c. The boron/aluminum wing box to be similar in overall size and dimensions to the graphite/epoxy wing box specimen in order to fit the existing test fixture.

Total length = 88.0 inches

Length of composite section = 52 inches

Distance between spars = 30 inches

Depth at wing spar = 5.50 inches

Depth at center = 7.50 inches

The wing cross-section geometry is shown in Figure 2.1.

2.3 STRUCTURAL DESCRIPTION

2.3.1 GENERAL. The wing box test specimen is composed of a composite center section 52 inches long with two steel load introduction fittings at each end, bringing the total length to 88 inches. The lower wing surface, including the access door, is illustrated in Figure 2.2.

5.6 Mil boron/6061 aluminum in the form of diffusion bonded sheets was utilized for the composite center section. This material was purchased to specification 72C0191 which was prepared for this program and which may be found in Appendix A.

2.3.2 SPARS. The wing spars are sandwich structures using $[\pm 45]_S$ boron/aluminum face sheets on 4.3 lb/ft^3, 3/8-inch thick aluminum honeycomb core. The boron/aluminum utilized in the spar webs is 5.6-mil boron/6061 aluminum The use of 4 plys of this material results in a web which is somewhat over strength for the applied loads. A web consisting of a $[\pm 45]$ skin balanced by a $[\mp 45]$ skin was considered

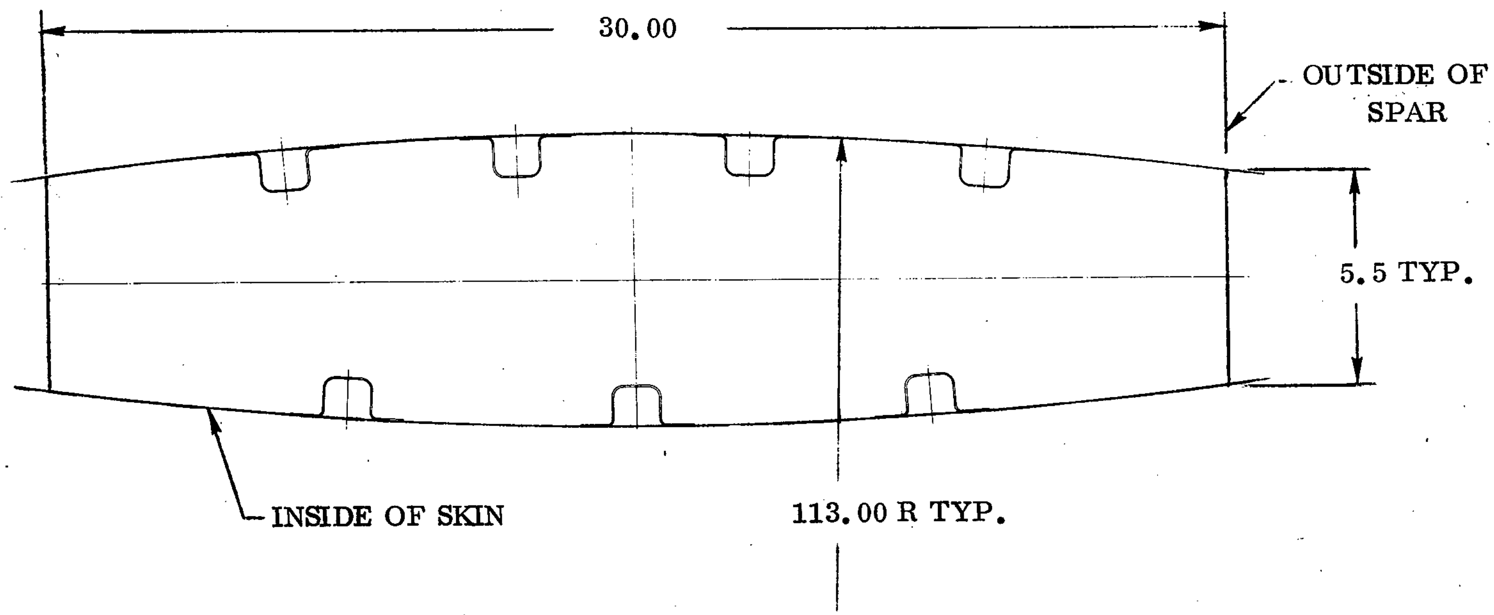


Figure 2.1. Wing Box Cross-section Geometry.

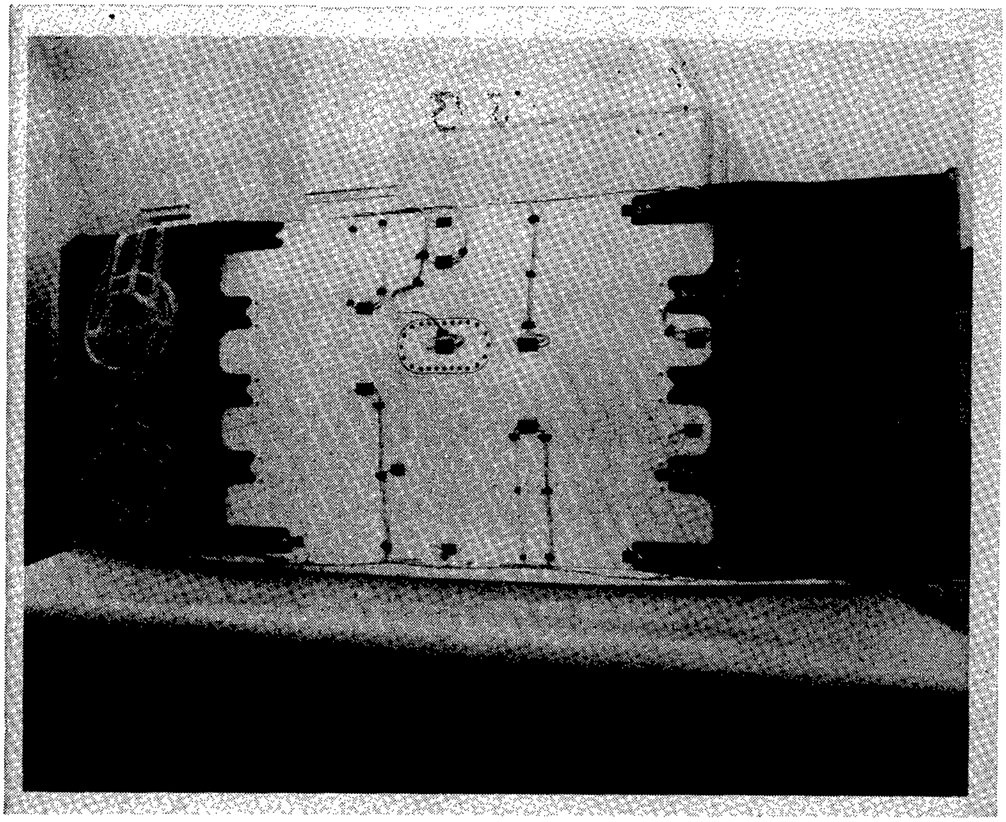


Figure 2.2. Lower Wing Surface, Including the Access Door.

but was discarded because the potential difficulty of successful assembling the spars with warped, unbalanced face sheets and the fact that this configuration would show a slightly negative margin of safety.

The spar caps each consist of two pieces of unidirectional boron/aluminum. The primary spar cap is a piece of $[0_8]$ flat sheet. A shaped filler of $[0_{29}]$ boron/aluminum is fitted into the space between the fillet radii at the two 0.032 thick titanium angles which attach the spar-caps to the web. (Figure 2.3.)

The spar was designed for assembly by means of adhesive bonding. The core is filled with epoxy potting locally in areas of load introduction; along the spar caps, ribs, and end load introduction fittings. This method, although heavier, was chosen over inserts as being simpler and more tolerant to mislocation of holes during assembly. Lockbolt fasteners are used to reinforce the attachment of the titanium angles to the spar web. The spar cap angles and spar cap are also attached mechanically by the rivets which attach the skin. Figure 2.4 shows the completed spar assemblies.

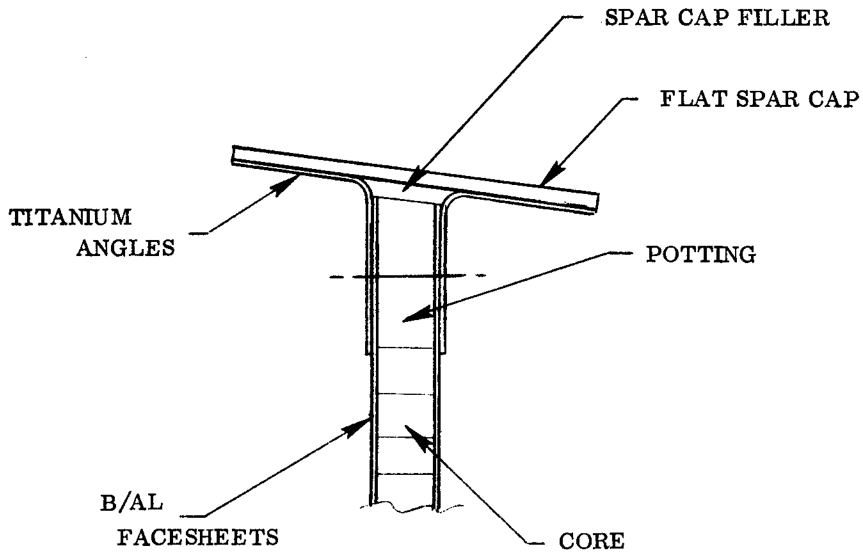


Figure 2.3. Spar Cap Details.

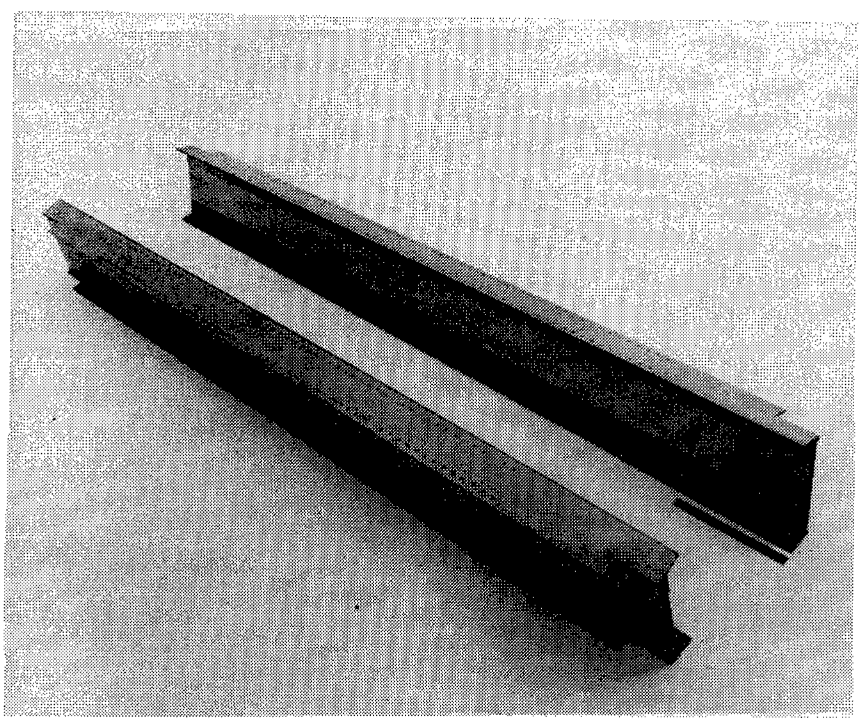


Figure 2.4. Completed Spar Assemblies.

2.3.3 <u>SKINS.</u> The upper and lower skins consist of $[\pm 45/\bar{0}_3]_S$ (7 ply) boron/aluminum sheet with the 0 plies oriented in the spanwise direction. The skins were obtained as flat sheets and wrapped to the contour over the ribs during assembly.

The skins were designed to buckle on the compression side in bending and to wrinkle in diagonal tension during torsional loading.

The skins were stiffened by $[0_6]$ boron/aluminum hat sections; 3 on the lower (tension) skin and 4 on the upper (compression) skin.

Figure 2.5 shows two of the stringers during fabrication.

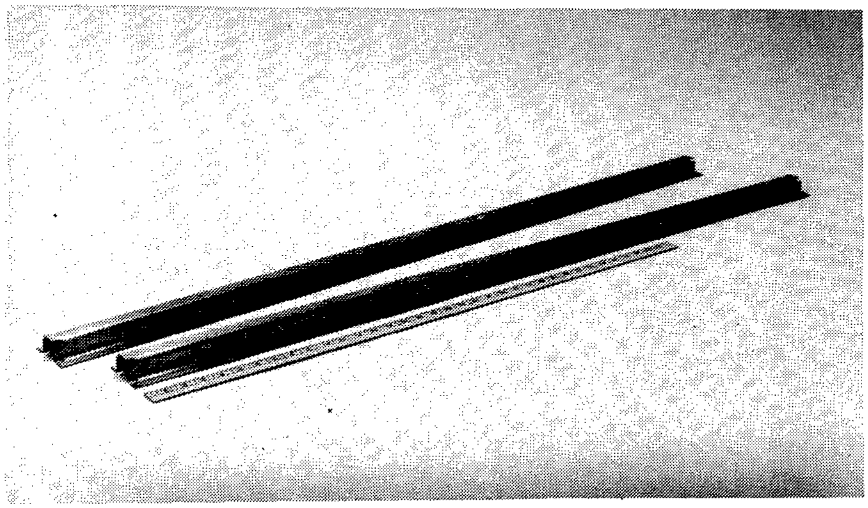


Figure 2.5. Formed Boron/Aluminum Stringers.

Initially two methods of stringer attachment were contemplated. It was known that spotwelding provides a high strength low cost method of assembly for boron/aluminum, while riveting is the conventional method of assembly for wing sections. Both of these methods were evaluated and crippling specimens of both types tested (Section 3). From a strength standpoint either method could be utilized. As the design progressed, however, it became apparent that due to the relatively small size of the specimen and the nature of the rib assemblies it would not be possible to use spotwelding. Spotwelding the stringers to the skin panels would result in skin-stringer subassemblies which must then be attached to the ribs and spars. The limited space within the box precluded the possibility of making adequate connections between the ribs and stringers. The riveted construction was therefore selected.

2.3.4 RIBS. The wing box has 4 ribs, two of which are located at the load introduction fittings. These ribs have aluminum caps, boron/aluminum webs and web stiffeners. The two center ribs have boron/aluminum rib caps and posts. All four ribs are attached to the spars and stringers by means of titanium angle brackets.

Figure 2.6 illustrates a typical boron/aluminum rib cap formed from a flat sheet of unidirectional boron/aluminum. This is the most complex part to get formed from sheet boron/aluminum.

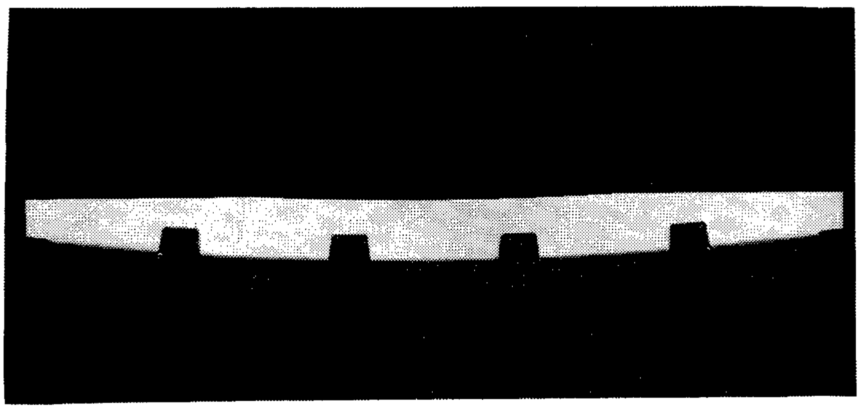


Figure 2.6. Completed Rib Cap Formed from Sheet B/Al.

2.3.5 LOAD INTRODUCTION AREAS. The load introduction areas of the wing box consist of steel plate structures which have the same cross section and contour as the composite center section. Load introduction into the composite section are by means of fingered splice plates. Spar cap loads, stringer and skin loads and spar web loads are transmitted by means of high strength steel tapered splice plates. The shape and taper of these members was selected after careful design studies to minimize load peaking.

Typical high fatigue life detail design techniques were utilized in the design of the steel to composite joints. The stringer termination and connection to the steel end fitting is shown in Figure 2.7. This drawing is of the stringer joint test and is typical of all of the 7 wing stringers. Shown are the tapered finger doublers and backup angles used to distribute load into the stringer. No adhesives were used in the load area; all loads being transmitted by mechanical fasteners.

Figure 2.8 illustrates the wing box during assembly. Shown are the spar splice plates and load introduction finger doublers.

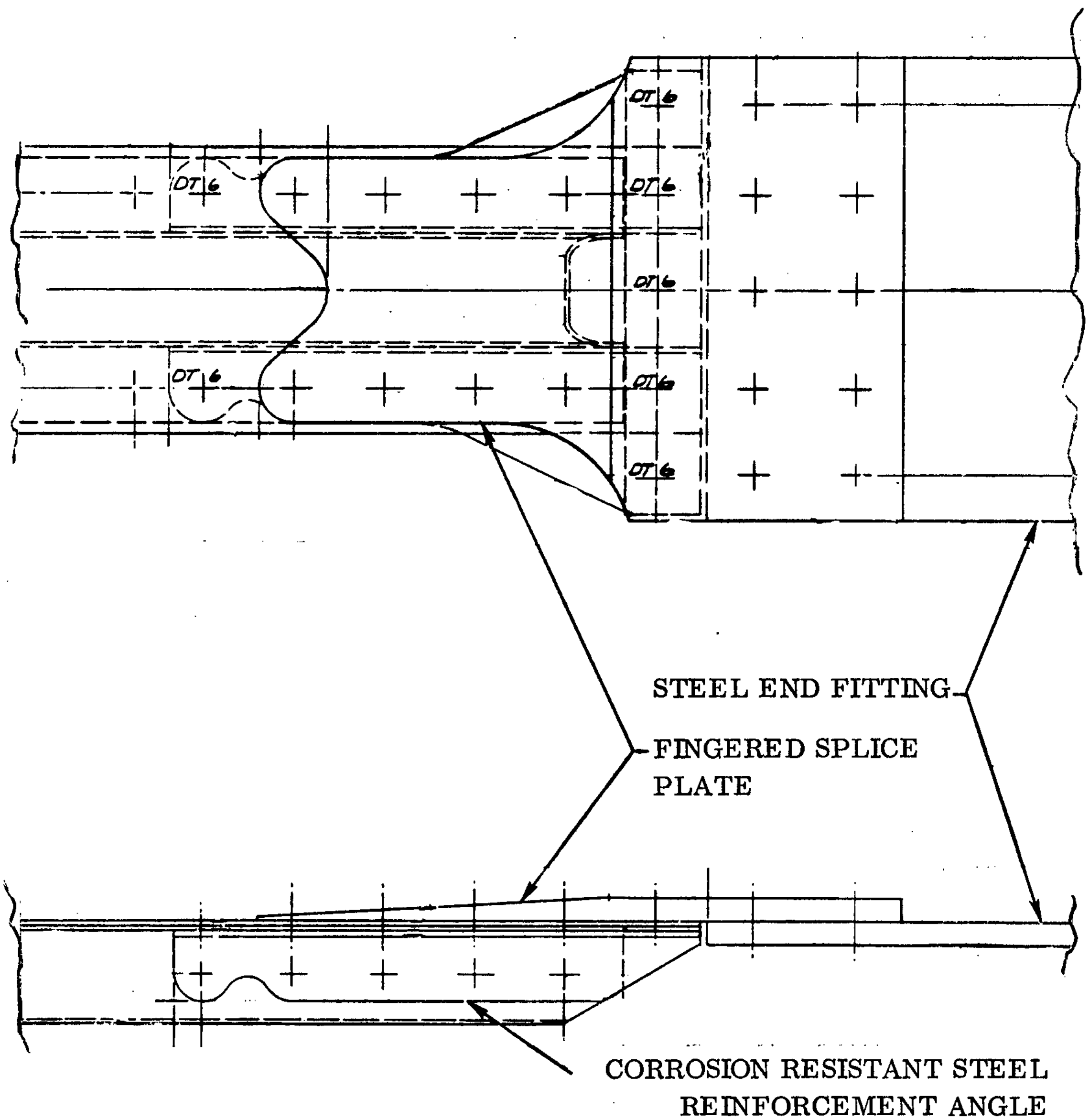


Figure 2.7. Detail of Stringer Load Introduction Method.

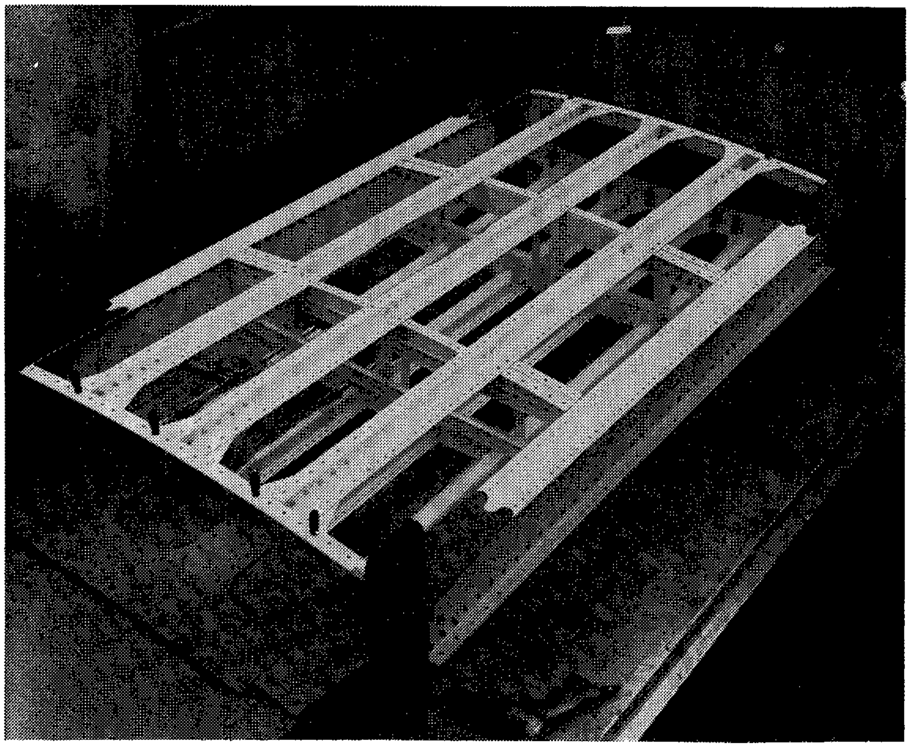


Figure 2.8. Wing Box during Fabrication.

Table 2-1 lists the engineering drawings which define the wing box configuration. The wing assembly drawing and other major subassembly drawings are found in Appendix B.

Table 2-1. Drawing List

Dwg. No.	C/L	Dwg. Title	No. Req'd.	N/A
72C0160	C	Boron Aluminum Wing Box - Test Specimen, Assy of	1	End Item
72C0161	—	Rib Cap - Boron Aluminum Wing Box, Test Specimen	2 of each	72C0160
72C0162	—	Access Door, Boron Aluminum Wing Box, Test Specimen	1	72C0160
72C0163	—	Doubler - Acess Door, Boron Aluminum Wing Box, Test Specimen	1	72C0160
72C0164	—	Filler - Access Door, Boron Aluminum Wing Box, Test Specimen	2	72C0160
72C0165	B	Stiffener - Hat, Assy of Boron Aluminum Wing Box, Test Specimen	7	72C0160
72C0166	A	Upr. Splice - Fingered, Boron Aluminum Wing Box, Test Specimen	2	72C0160
72C0167	A	Lwr. Splice, Fingered, Boron Aluminum Wing Box, Test Specimen	2	72C0160
72C0168	—	Angle Bracket - Boron Aluminum Wing Box, Test Specimen	2	72C0160
72C0169	—	Lug Fitting, Boron Aluminum Wing Box, Test Specimen	4	72C0160
72C0170	—	Splice Plate, Boron Aluminum Wing Box, Test Specimen	4 ea.	72C0160

Table 2-1. Drawing List (Continued)

Dwg. No.	C/L	Dwg. Title	No. Req'd.	N/A
72C0171	B	End Supt. - Assy of Boron Aluminum Wing Box, Test Specimen	2	72C0160
72C0172	A	Compression Specimen, Hat Stiffener, Boron Aluminum	6	End Item
72C0173	C	Compression Specimen, Splice Section, Boron Aluminum - Steel Supt.	2	End Item
72C0174	B	Tension Specimen - Splice Section Boron Aluminum - Steel Supt.	1	End Item
72C0175	—	Lug Fitting - Tension Specimen	1	72C0174
72C0176	—	Splice Plate - Spar Web, Boron Aluminum Wing Box, Test Specimen	4	72C0177
72C0177	A	Spar - Assy of Boron Aluminum Wing Box, Test Specimen	2	72C0160
72C0178	—	Support Brackets - Titanium, Boron Aluminum Wing Box, Test Specimen		72C0160
72C0179	A	Rib Cap - Boron Aluminum Wing Box, Test Specimen	2	72C0160
72C0183	A	Fillers - Boron Aluminum Wing Box, Test Specimen		72C0160
72C0184	—	Shear Web - Boron Aluminum Wing Box, Test Specimen		

2.4 STRESS ANALYSIS

2.4.1 INTERNAL LOADS. The internal loads, shear flows and member stresses were determined by computer program GENFETE.

The structural model is defined by a composite of axial load carrying "truss" members connected to shear carrying panels. The model simulates a conventional wing box with a front and rear spar, and an upper and lower wing cover stiffened by stringers (4 on the upper compression surface and 3 on the lower tension surface). The box is symmetrical in the chordwise and spanwise direction. The vertical direction is unbalanced because of one stringer more on the upper surface. The model box beam is loaded and/or stabilized by six chordwise frames. The composite structure is contained in three bays in the center.

The structural arrangement of the model finite elements is shown in Figure 2.9. The area or thickness of the webs and bars is shown in Figures 2.10 and 2.11.

The three primary loading condition applied loads and constraints are shown in Figures 2.12 through 2.15. The internal loads and shears for the 3 loading conditions are shown in Figures 2.16 through 2.30.

In addition to the data displayed here, GENFETE also calculates the deflection of each node point for each loading condition. Predicted deflection data is compared to test data in Section 5.

2.4.2 MATERIAL PROPERTIES AND ALLOWABLES. The following properties were utilized in the loads analysis:

Steel Ends

ASTM — A7 Steel Covers

$E = 30 \times 10^6$ psi

$G = 11 \times 10^6$ psi

Splice Plate
4130 CMS

$E = 30 \times 10^6$ psi

$G = 11 \times 10^6$ psi

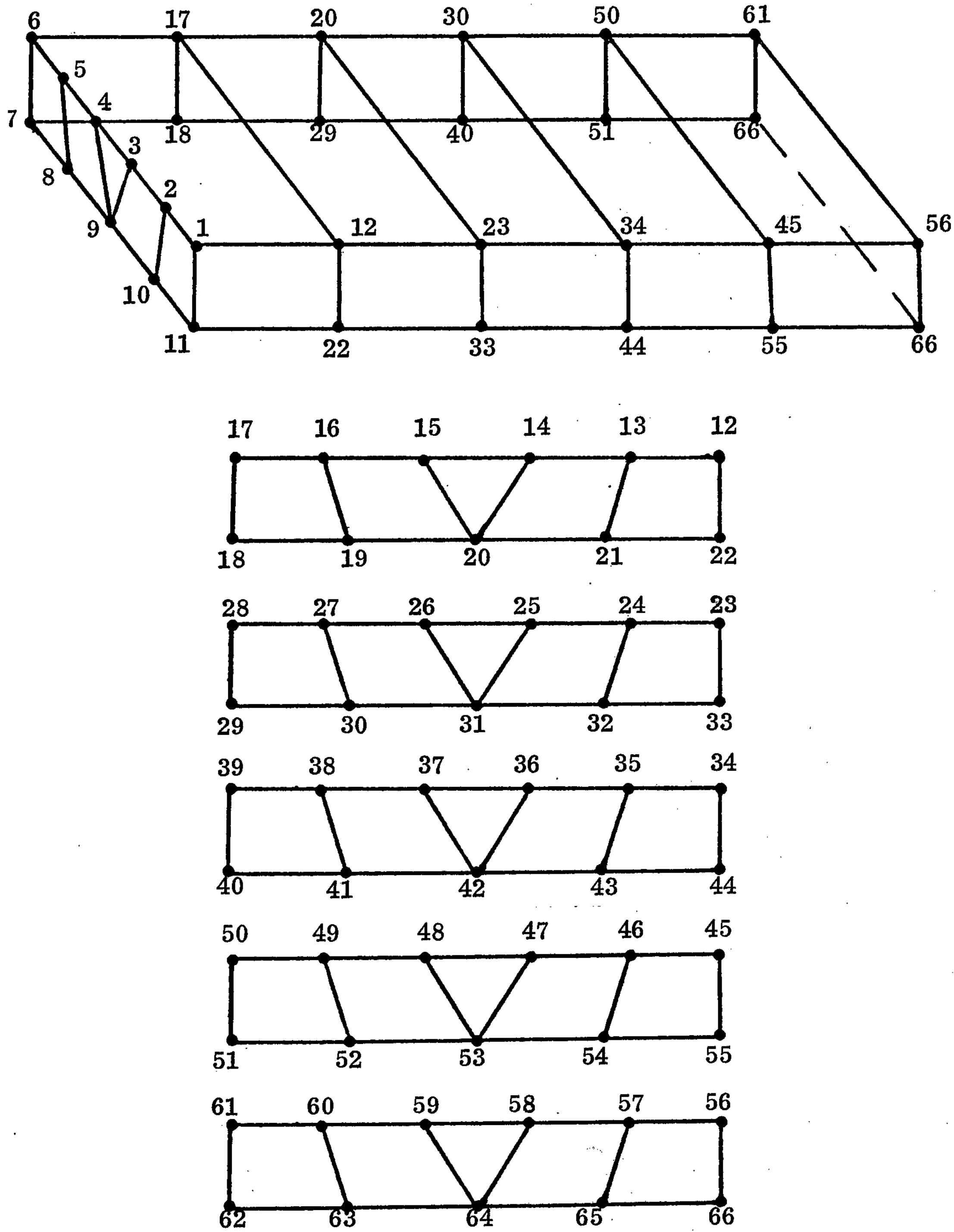
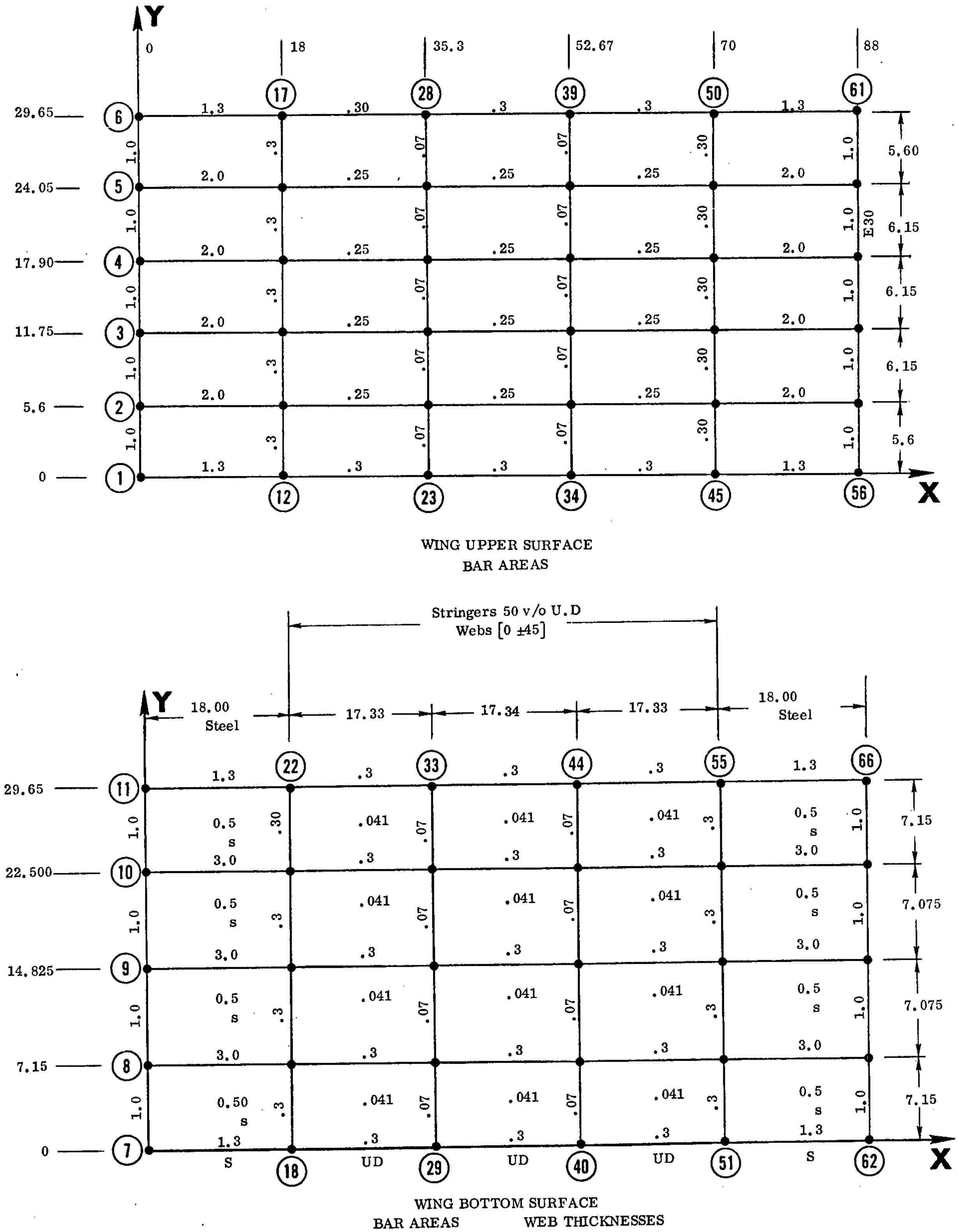
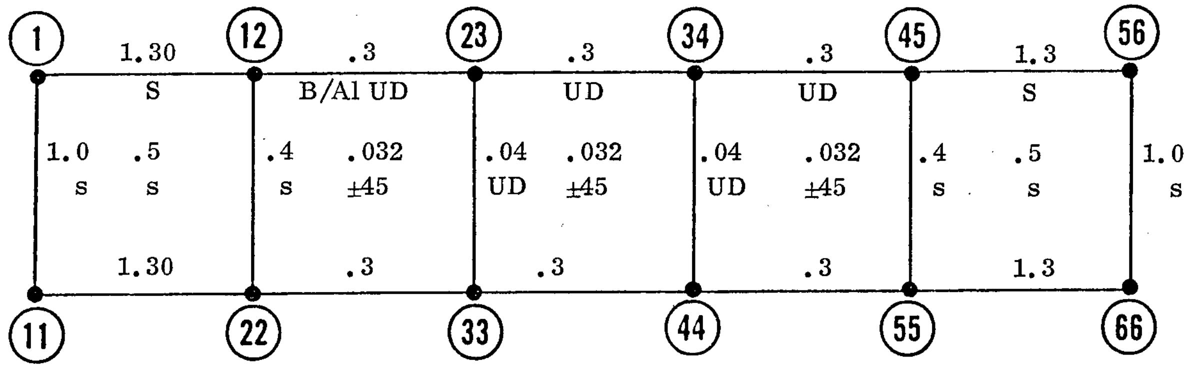


Figure 2.9. Structural Model Node Point Designations.

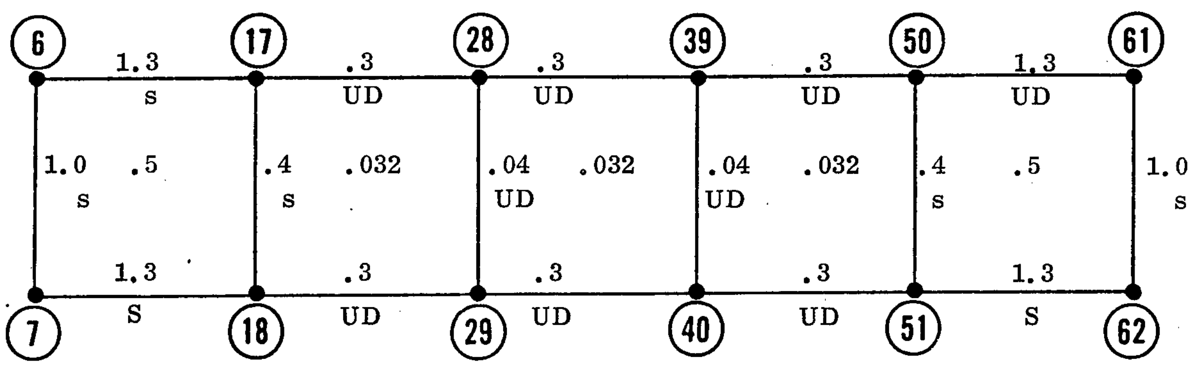


WING UPPER SURFACE
BAR AREAS
Stringers 50 v/o U.D
Webs [0 ±45]
WING BOTTOM SURFACE
BAR AREAS
WEB THICKNESSES

Figure 2.10.



REAR SPAR



FRONT SPAR

SPAR TRUSS AREAS AND WEB THICKNESSES
in^2/inches

Figure 2.11.

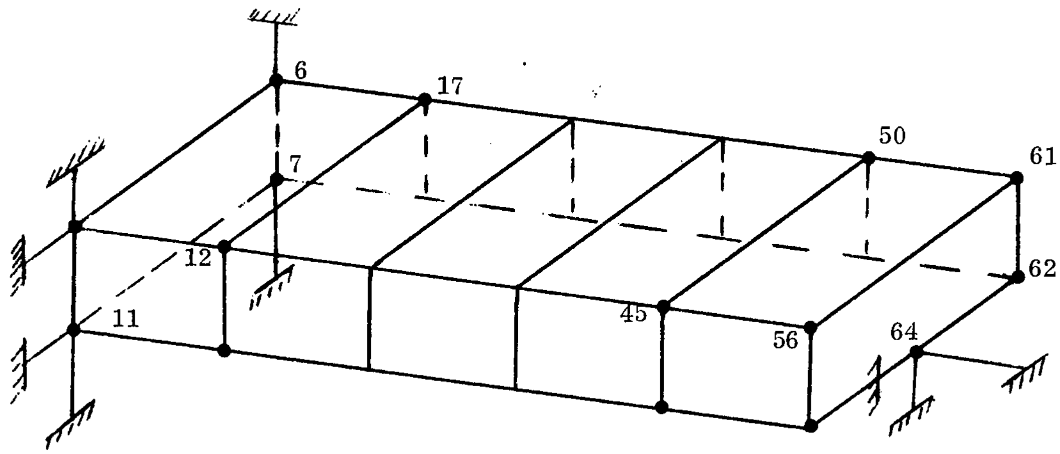


Figure 2.12. Structural Model Constraints (9).

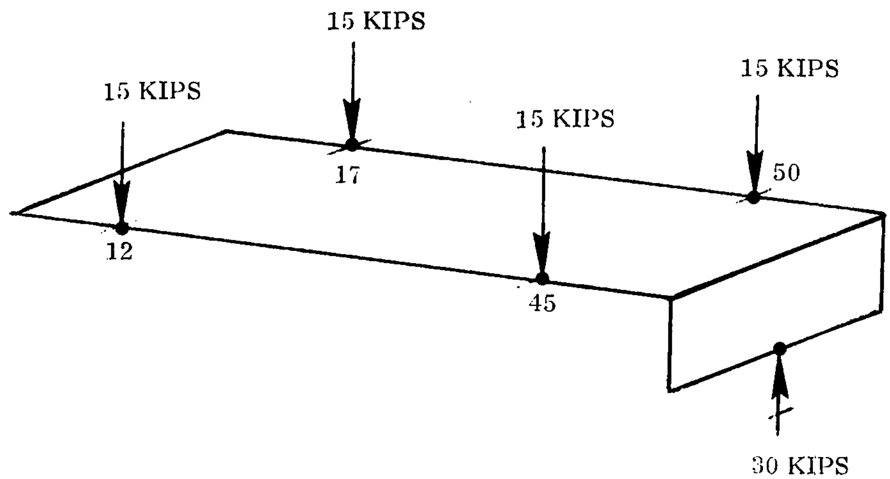


Figure 2.13. Case 1 Maximum Bending Loading Condition — 540,000 in-lb ult.

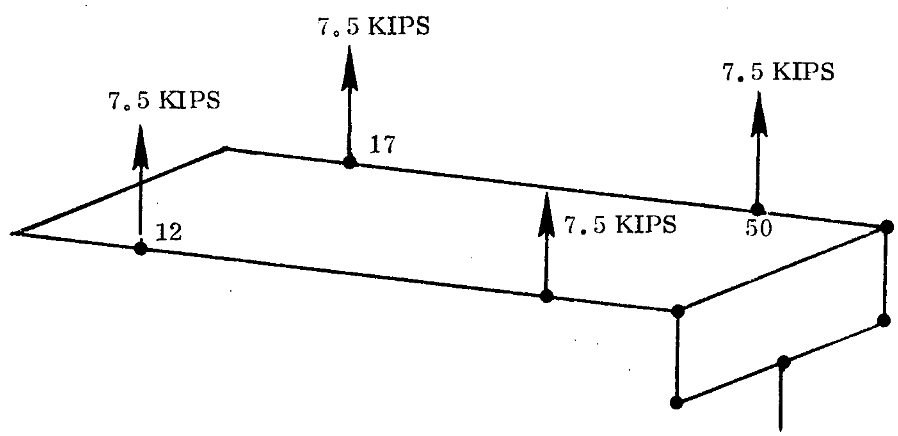


Figure 2.14. Negative Bending Loading Condition — 279,000 in-lb ult.

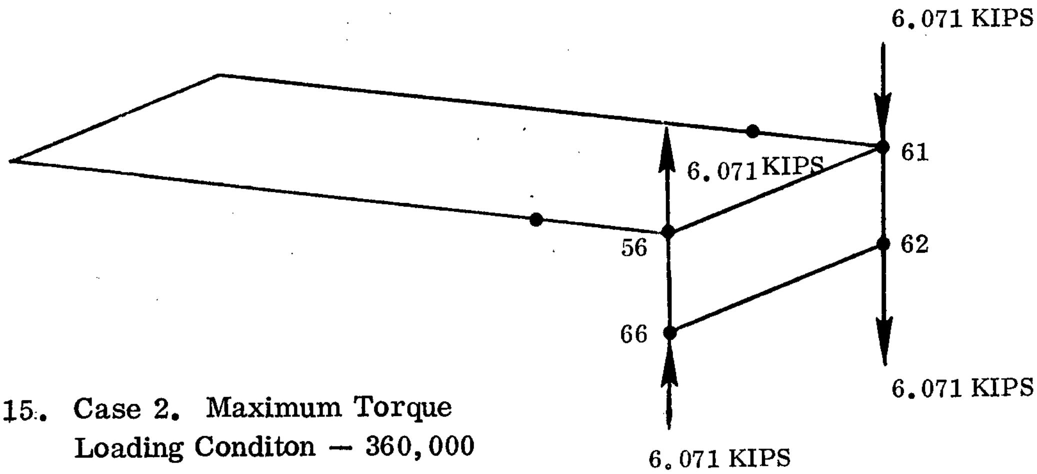


Figure 2.15. Case 2. Maximum Torque Loading Conditon — 360,000 in-lb ult.

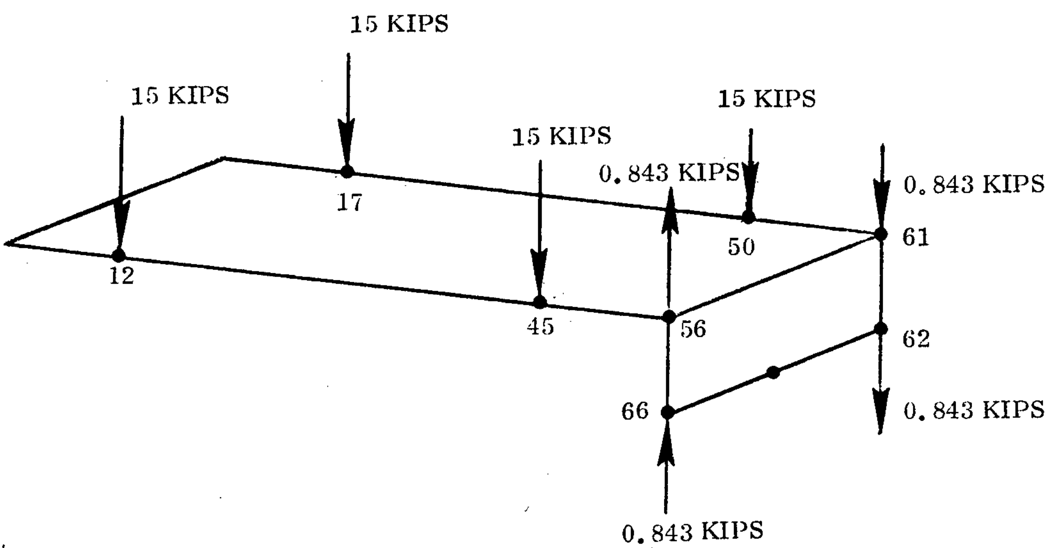
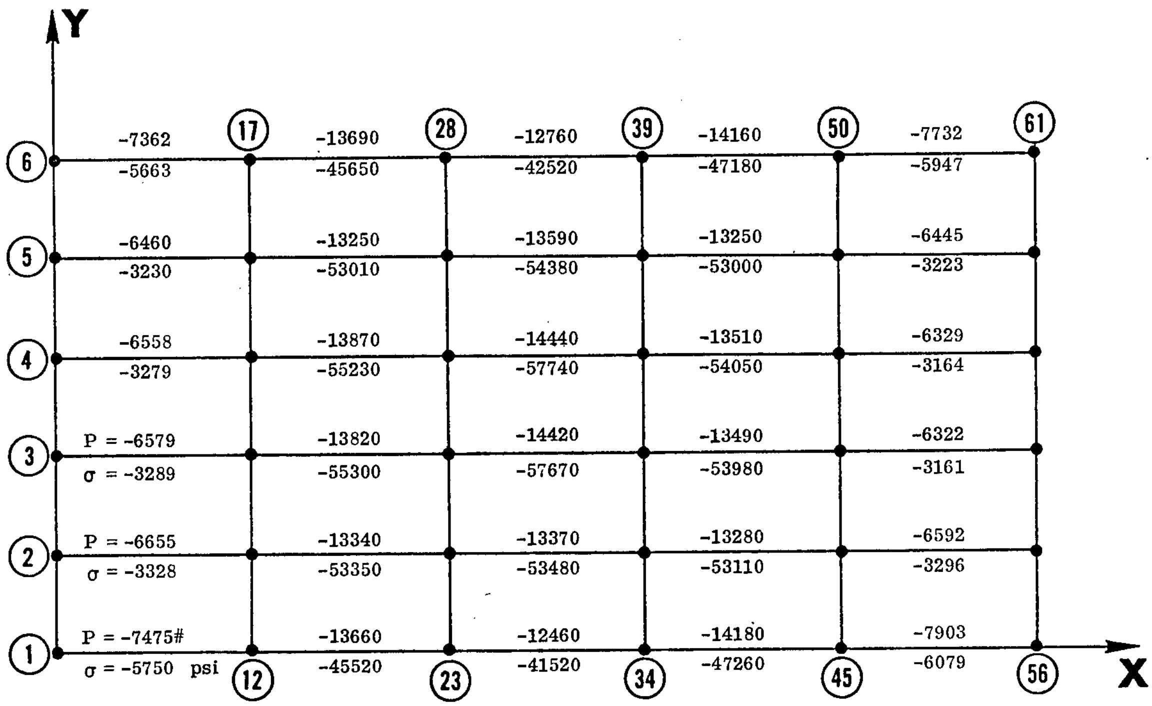
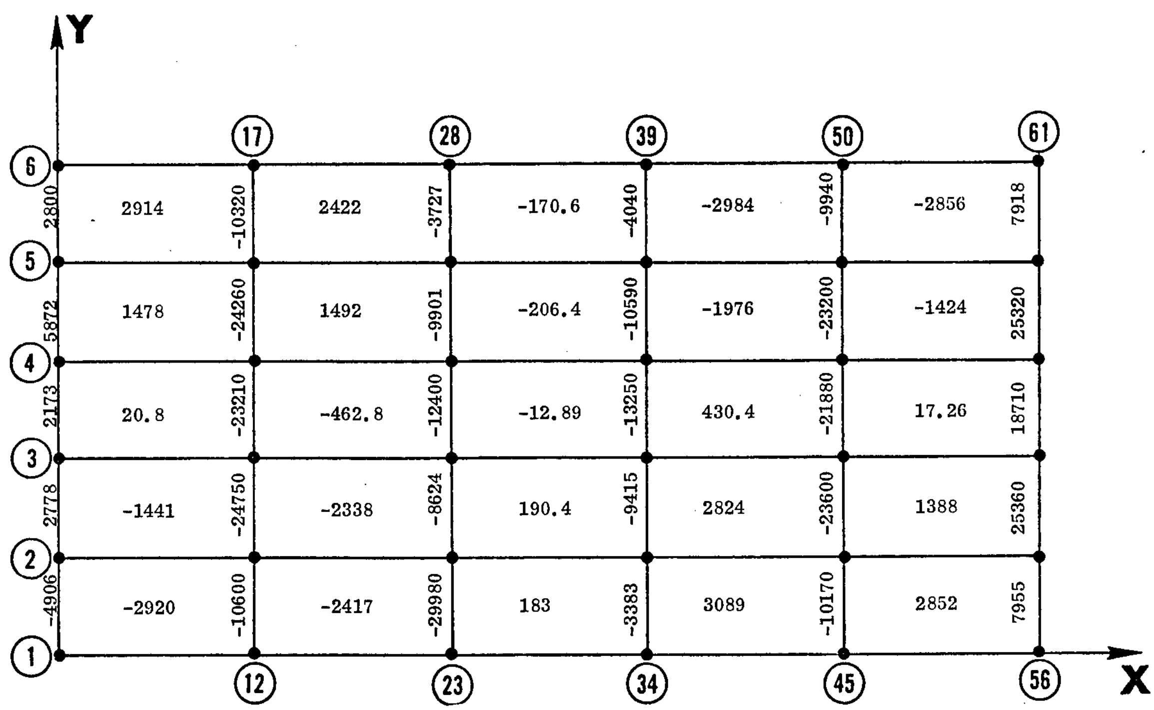


Figure 2.16. Case 3. Maximum Combined Loading Condition. M = 540,000 in-lb ult. T = 50,000 in-lb ult.



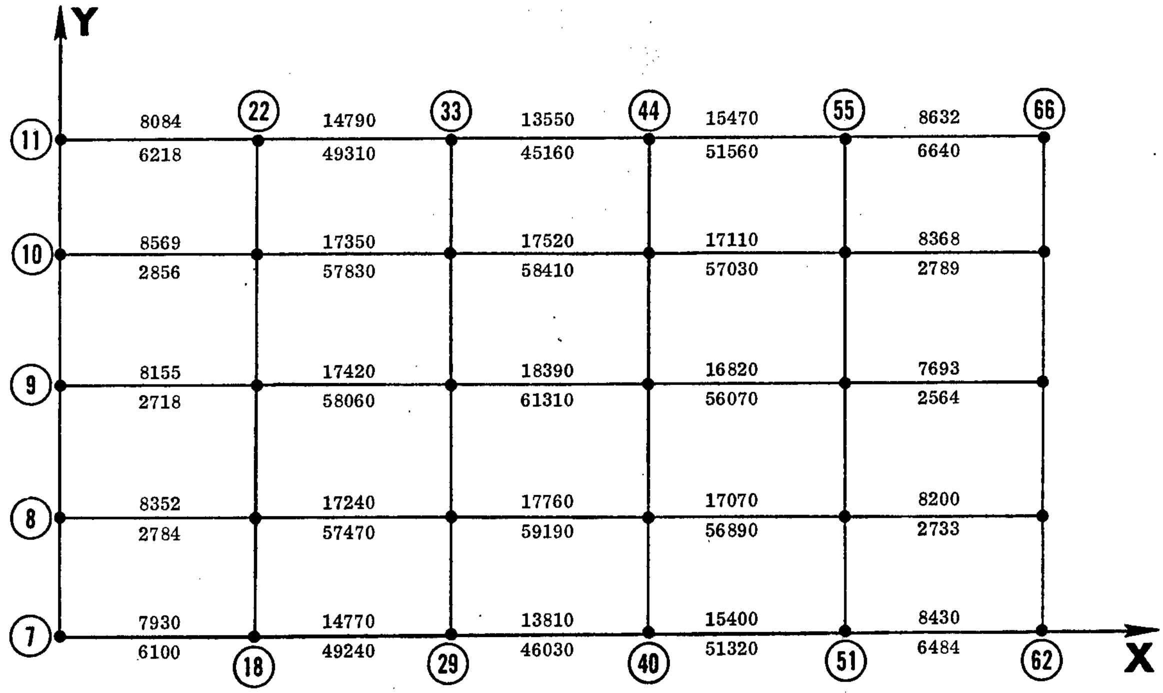
Wing Top Surface - Case 1 Maximum Bending
Stringers and Spar Caps - Loads and Stresses - Ultimate

Figure 2.17.



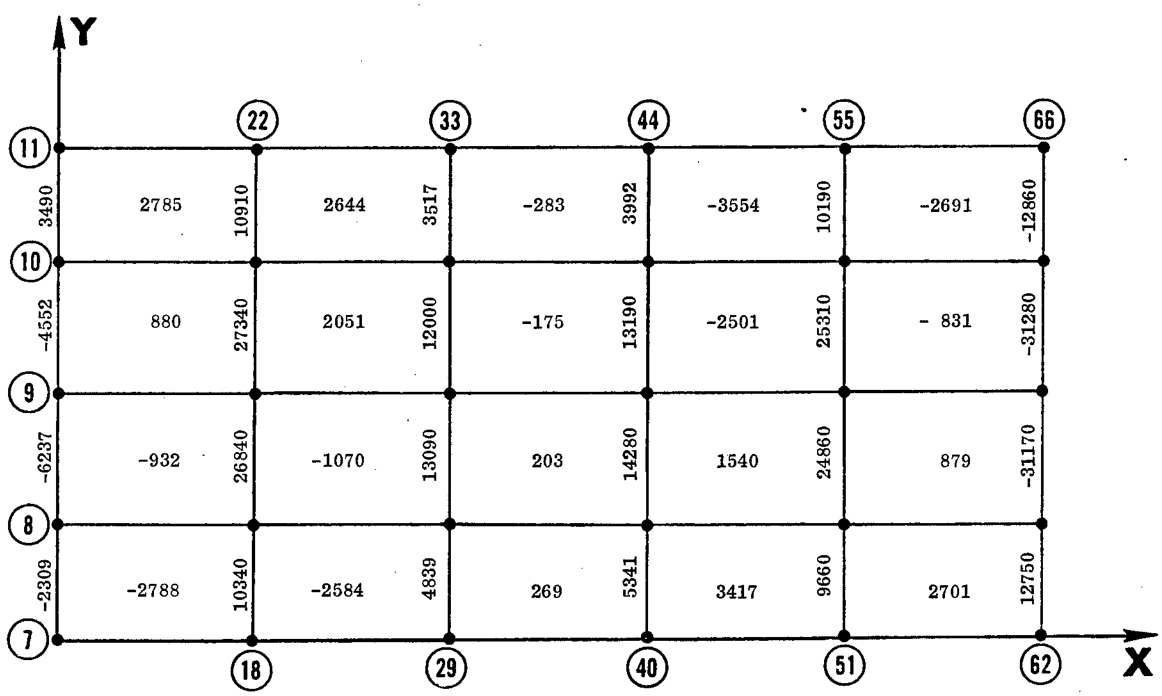
Wing Top Surface - Case 1 Maximum Bending
Web Stresses and Frame Cap Stresses psi -

Figure 2.18.



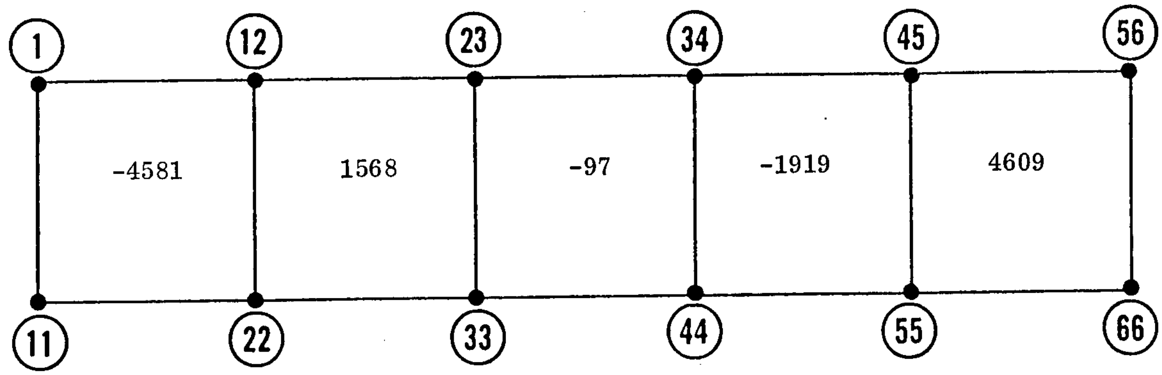
Wing Bottom Surface Case 1 Maximum Bending
Stringers and Spar Caps - Loads and Stresses -

Figure 2.19.

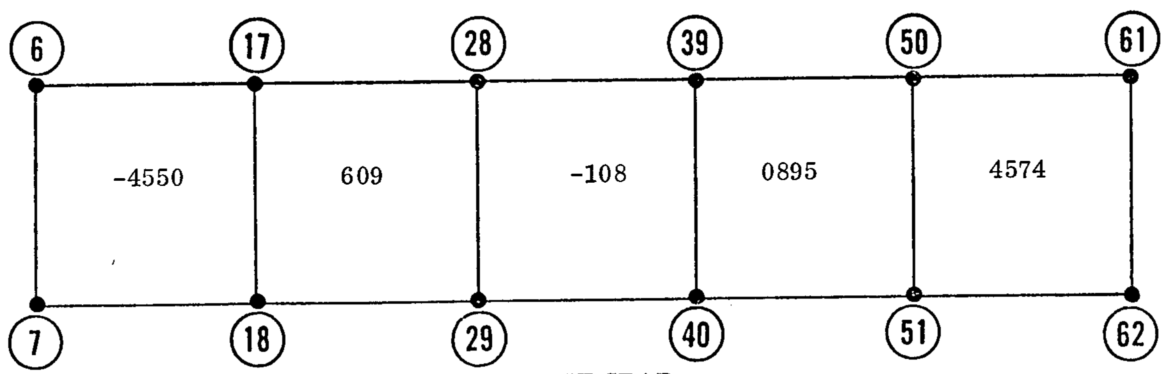


Wing Bottom Surface Case 1 Maximum Bending
Web Stresses & Frame Cap Stresses - psi -

Figure 2.20.



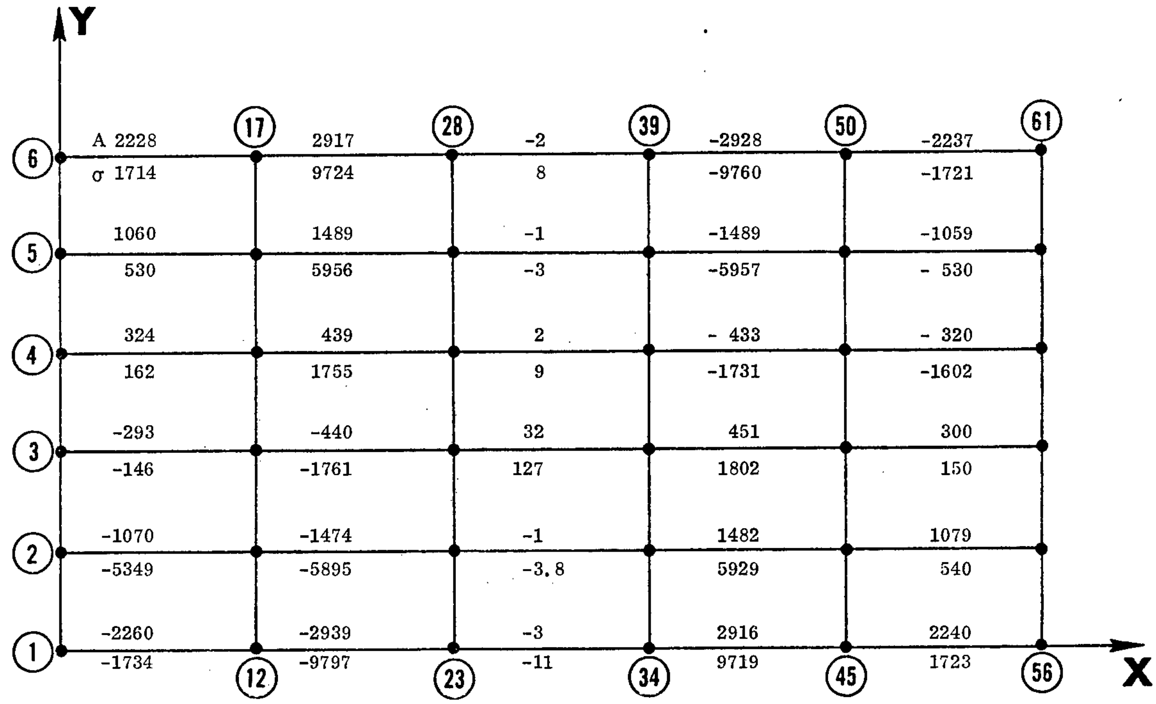
REAR SPAR



FRONT SPAR

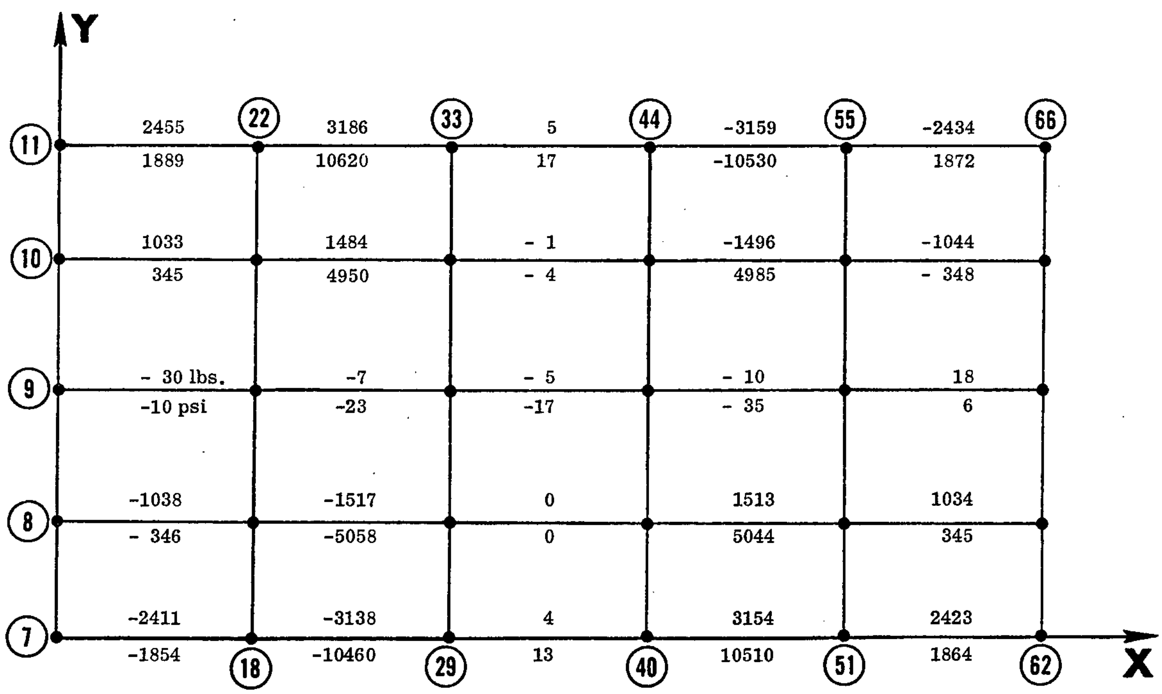
Wing Spars Case 1 Maximum Bending
Web Stresses - Ultimate

Figure 2.21.



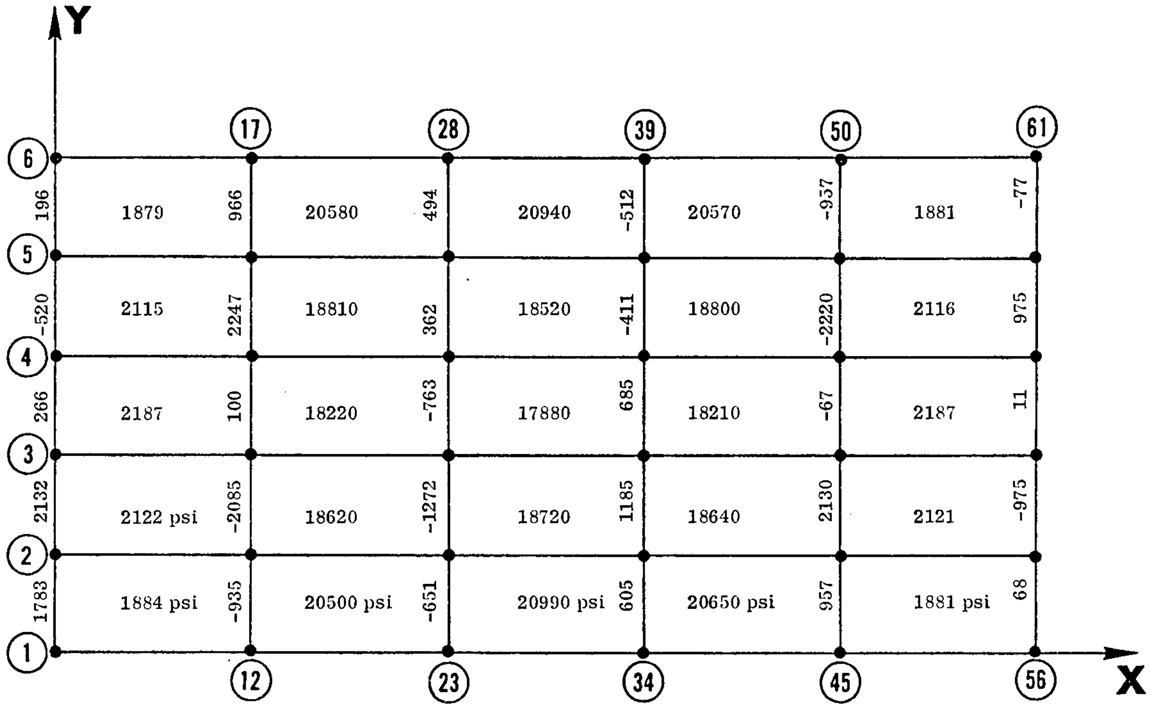
Wing Top Surface Case 2 - Maximum Torque
Stringers and Spar Caps — Loads and Stresses - Ultimate

Figure 2.22.



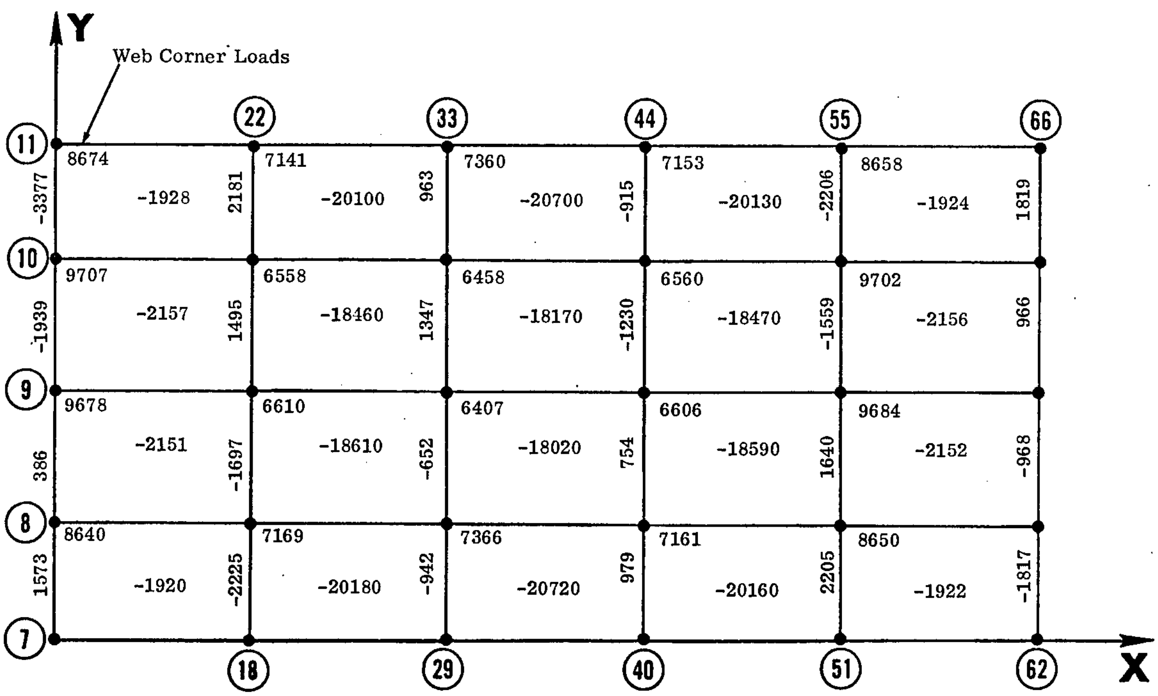
Wing Bottom Surface Case 2 - Maximum Torque
Stringers and Spar Caps - Loads and Stresses - Ultimate

Figure 2.23.



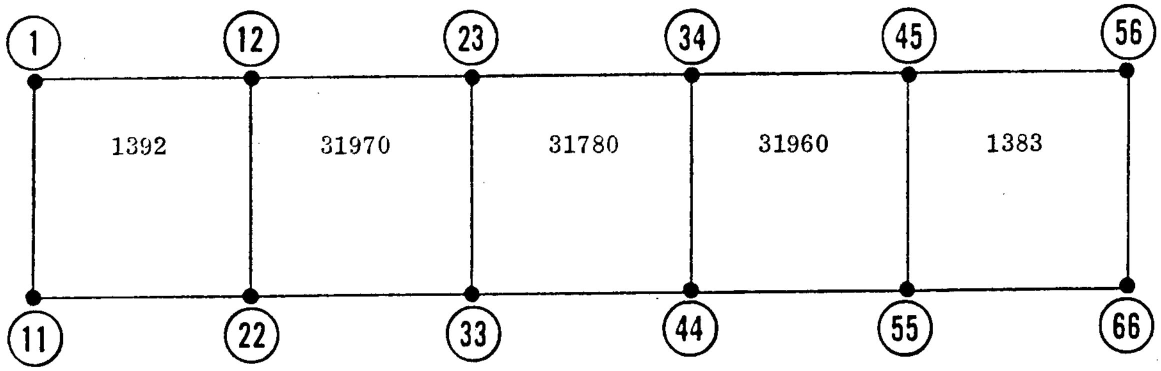
Wing Top Surface Case 2 - Maximum Torque
Web Stresses and Frame Cap Stresses - Ultimate.

Figure 2.24.

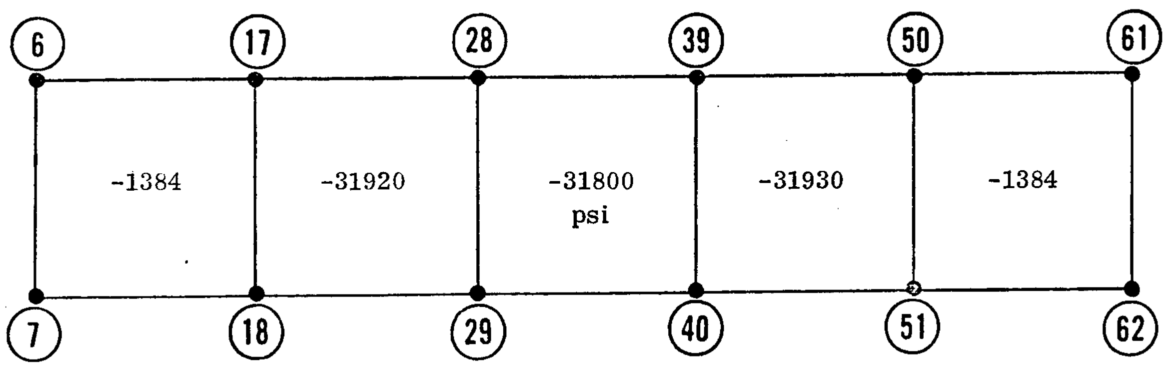


Wing Bottom Surface Case 2 - Maximum Torque
Web Stresses and Frame Cap Stresses - Ultimate

Figure 2.25.



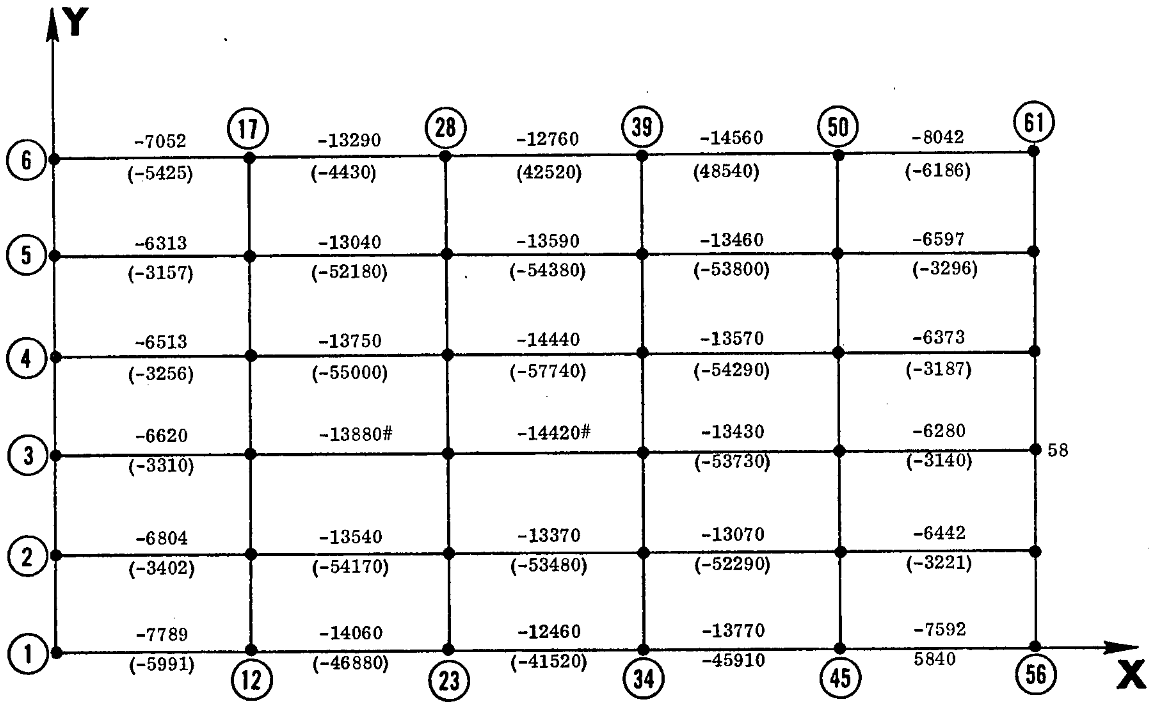
REAR SPAR



FRONT SPAR

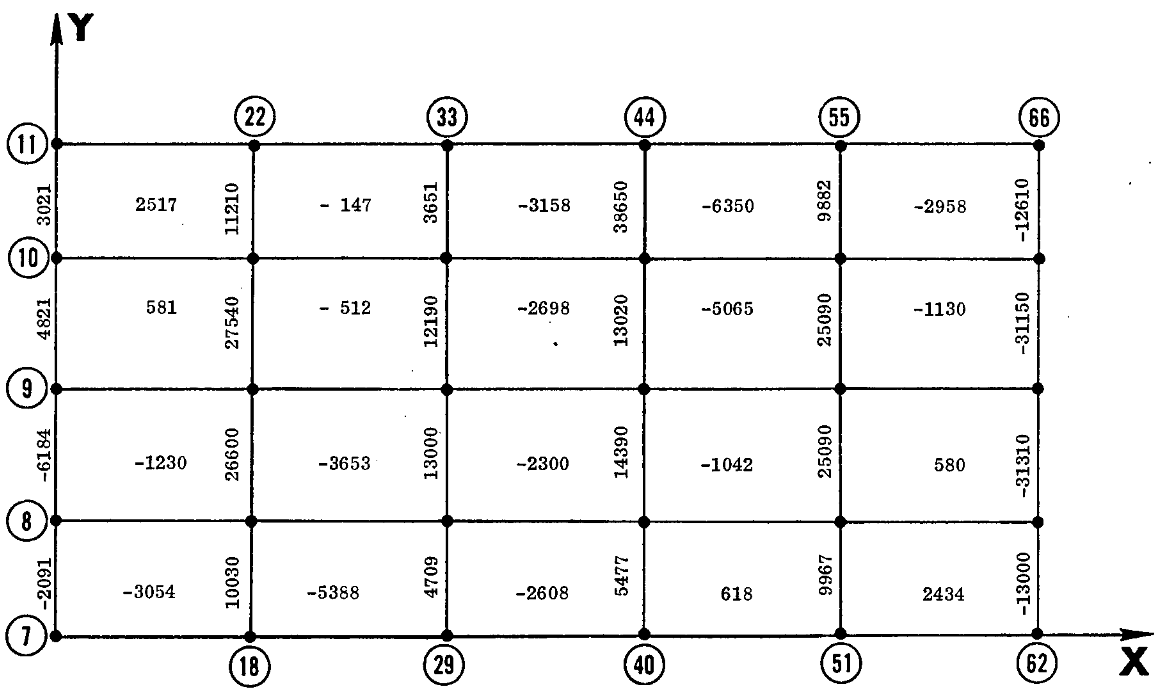
Wing Spars Case 2 - Maximum Torque
Web Stresses - Ultimate

Figure 2.26.



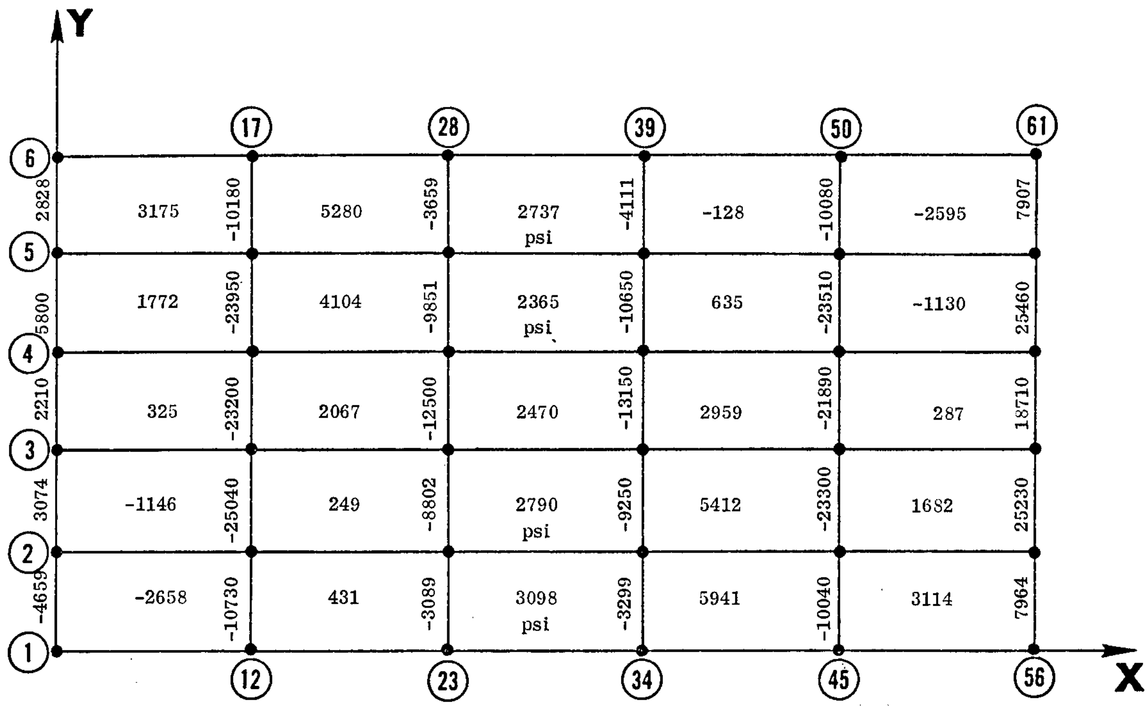
Wing Top Surface Case 3 Max. Bending + Torque
Stringer and Spar Caps - Loads and Stresses - Ultimate

Figure 2.27.



Wing Bottom Surface Case 3 - Bending + Torque
Web Stresses and Frame Cap Stresses - Ultimate

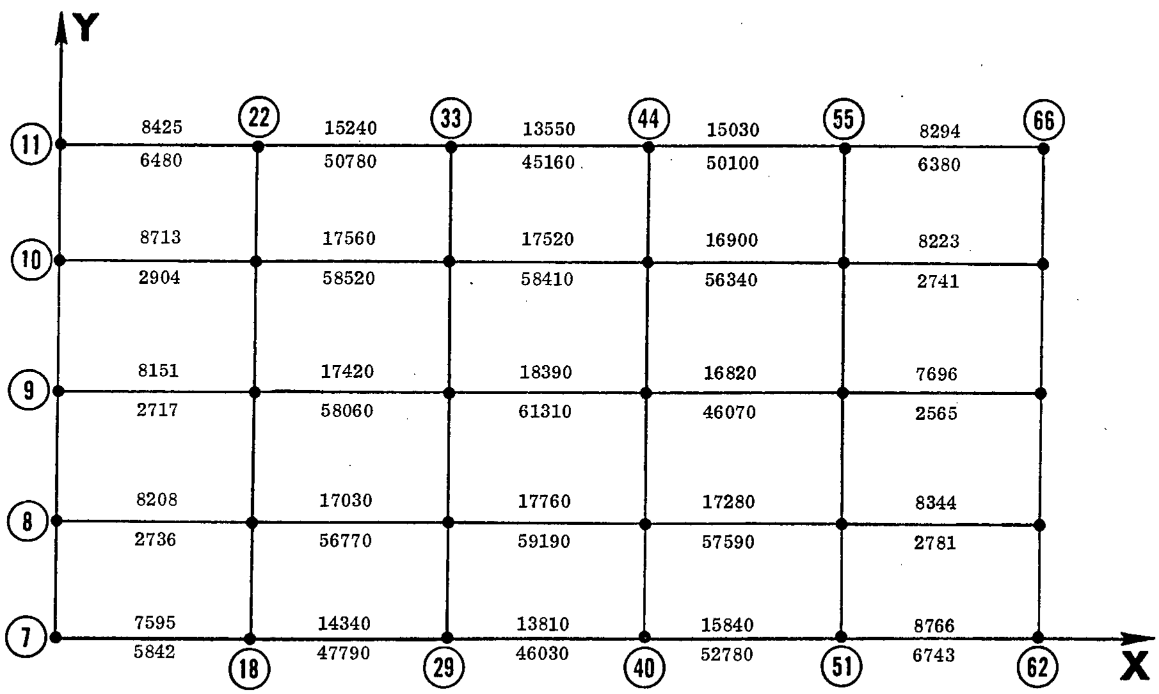
Figure 2.28.



Wing Top Surface - Case 3 Bending and Torque

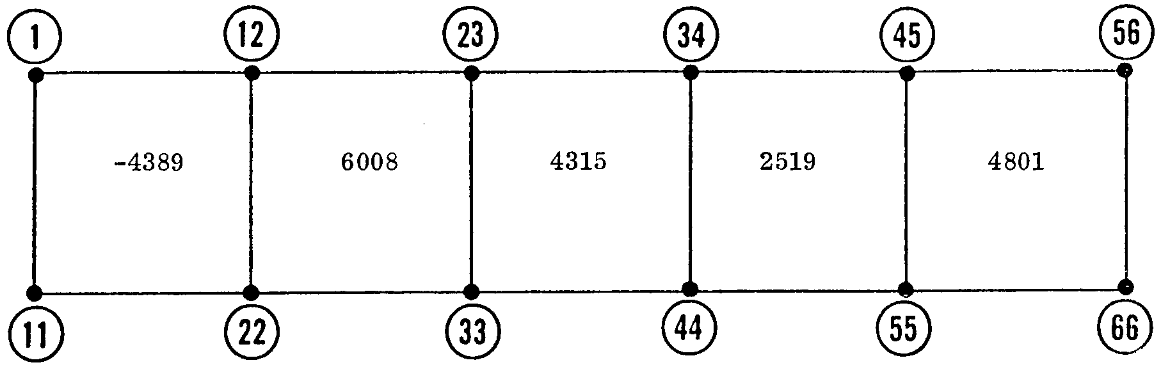
Web Stresses and Frame Cap Stresses Ultimate

Figure 2.29.

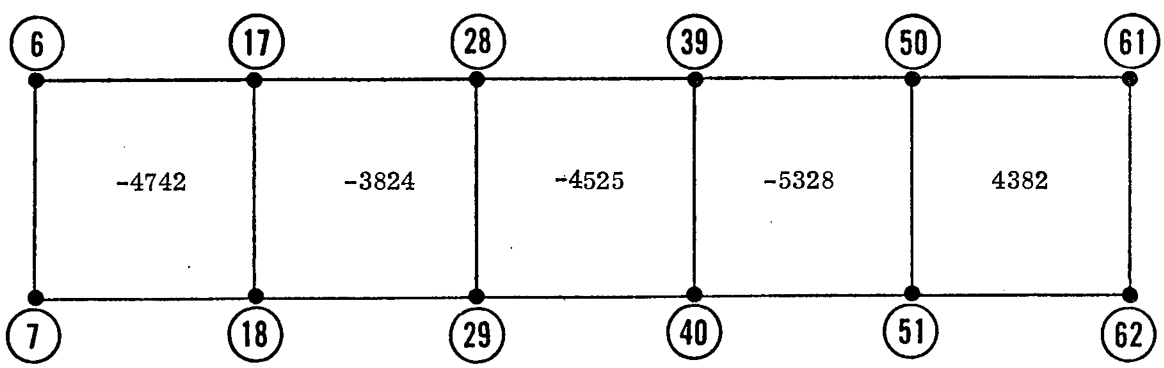


Wing Bottom Surface Case 3 - Bending + Torque
Stringers and Spar Caps — Loads and Stresses - Ultimate

Figure 2.30.



REAR SPAR



FRONT SPAR

Wing Spars Case 3 - Maximum Moment
+ Torque Web Stresses.

Figure 2.31.

Test Section — Boron/Aluminum

Basic Lamina U.D. 50 v/o (5.6-mil)

Stringer and Caps

$$E_1 = 29.0 \times 10^6 \text{ psi}$$

Cover Webs (0/±45)

$$G = 9.5 \times 10^6 \text{ psi}$$

Spars and Frames (±45)

$$G = 10.5 \times 10^6 \text{ psi}$$

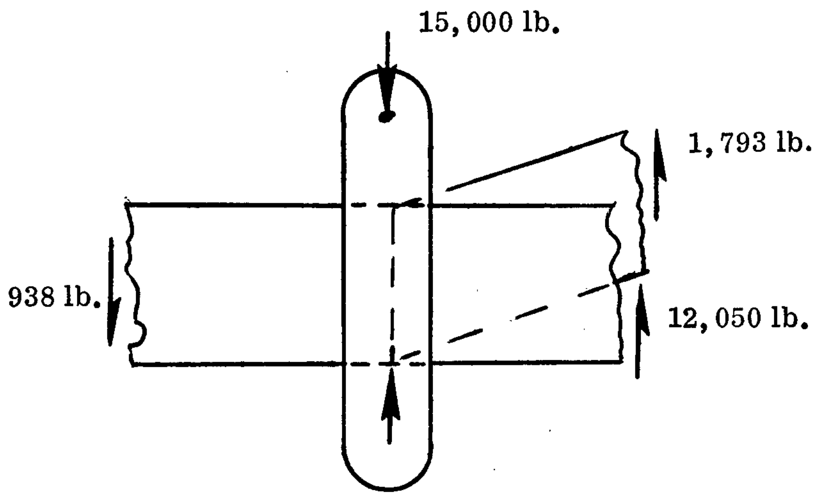
The allowables for conventional materials were obtained from MIL-HDBK-5. The design allowables for boron/aluminum are summarized in Table 2-2.

Table 2-2. Room Temperature Allowables for 5.6-mil Boron-6061 Aluminum

	Average	3 Sigma
F_{tu_1}	185 ksi	170 ksi
F_{tu_2}	12.7 ksi	8.7 ksi
E_{t_1}	31.8×10^6 psi	29.4×10^6 psi
E_{t_2}	19.1×10^6 psi	
F_{c_1}	226.7 ksi	200 ksi
F_{c_2}	24.2 ksi	
E_{c_1}	28.4×10^6 psi	
F_{su}	21 ksi	
G	8.3×10^6 psi	
ν_{12}	0.31	
ν_{21}	0.26	

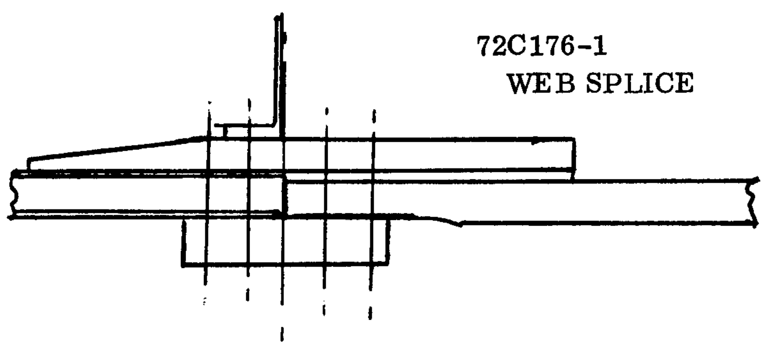
Reference: Report GDCA DBG 73-006 & AFML-TR-71-186.

2.4.3 ANALYSIS.



Spar-Frame-Fitting Intersection

ΣZ = +12,050 + 1,793 +938 = 14,781 vs 15,000. The rib caps and the cover-corner loads produces 77 +142 = 219 pounds which accounts for difference.



The 12,050 pounds will be checked for attachment to the fitting by 7 1/4 inch bolts - A286.

$$P_s = 4,470 \text{ lbs/bolt}$$

$$P = 4,470 \times 7 = 31,290 \text{ lbs.}$$

The 12,050 load reacts all but 2,950 pounds of the fitting load. This 2,950 is assumed reacted by 938 pounds from the test section web and 2,012 pounds from the frame. The frame loading is transmitted through one row of 5-1/4 inch bolts.

$P_s = 5 \times 4,470 = 22,350$ lb.

Bearing on the .032 sandwich is critical.

$P_{brg} = 100,000 \times .032 = 3,200$ lb.

Reference 72C0169 for fastener pattern.

7 NAS 1004 bolts through support spar web
ASTM A7 $t = .312$

8 NAS 1004 bolts through spar splice web
72C0176-1 .125 CMS HT 125

A286 bolts $F_{tu} = 140$ ksi

$D = .274$ $F_{su} = 91$ ksi MIL-HDBK 5A

$P_s = 4,470$ lbs/bolt

For A7, $t = .25$

$$P_{brg} = \frac{4,470}{.25 \times .25} = 71,500 \text{ psi } \underline{\text{OK}}$$

For 1/8 CMS

$$P_{brg} = \frac{4,470}{.25 \times .125} = 143 \text{ ksi } \underline{\text{OK}}$$

<u>Attachment of Load Fitting</u> 72C0169 — 72C0169 1/2 inch ASTM-A7 steel

<u>Case 1 &</u> 3

$P_z = 15,000$ lb.

$$M.S. = \frac{15 \times 4,470}{15,000} - 1 = \underline{3.47}$$

Access Door

Door 72C0162
72C0163 at panel 30, 41, 42, 31 or 31, 42, 43, 32
Doubler 72C0164

Critical for shears +10 psi loads.

Case 1

$q = -203$

Case 2

$f_s = -18,170$ psi

Case 3

$f_s = -2,698$ psi

Hoop Tension 10 psi ultimate

$f_{ty} = 27,500$ psi

Chordwise loading $N_y = 27,500 \times .041 = 1,130$ lb/in (max)

Spanwise loading $N_x = 34.5$ lbs/in

72C0163 is a $(\pm 45/0_2)_s$

An opening 3.35 inches wide is reinforced with a 6.12 inch doubler and an .041 door having a thickened edge for fasteners.

Chordwise attachment is by 5 NAS 1153C3 screws through $2 \times .05 = .10$

A286 screw 10-32 $F_{su} = 91$ ksi

$P_s = 507 \times 1.5 = 1.265$ lb/in in .100

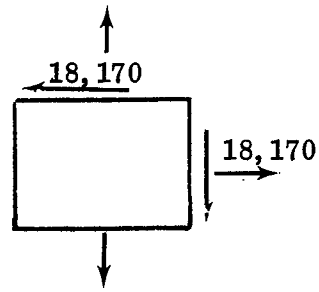
$Q = 5 \times 760/3.35 = 1,890$ lb/in

$1,130/.05 = 37,800$ psi

$f_b = 1,890/(2 \times .05 \times .188) = 65.5$ ksi

Case 2 has maximum shear stress

$f_s \quad = \quad 18,170$ psi



adding a longitudinal tension due to 10 psi pressure.

$f_t \quad = \quad 34.5/.05 = 690$ psi

$f'_s \quad = \quad (18,170^2 + 690^2)^{1/2} =$

$18,183$ psi

$$M.S. = \frac{22,600}{18,183} - 1 = \underline{.24}$$

Case 1 + 10 psi

$f_t \quad = \quad 61,310$ psi El. 31-42 with no effective skin

$A_{str} \quad = \quad .3 \text{ in}^2$

$A_{sk} \quad = \quad .05 \times 7.075 = .354$

For skin effective

$$P_{skin} = \frac{61,310 \times .3 \times .354 \times 25.5}{.3 \times 29 + .354 \times 25.5}$$

$= 9,350$ lbs/stringer

$q \quad = \quad 9,350/7.075 = 1,325$ lb/in

$1,890/1,325 - 1 = +.43$

Skins

The bottom skin is in tension for the maximum "X" cover loading which occurs in Case 3 Max Moment + Torsion.

The skin must have compatible longitudinal strain with the stringers.

Stringer 31-42 has a $f_t = 61,296$ psi

Skin panel 31, 42, 43, 32 has a shear stress of $\tau_s = -2,720$ psi

If the skin area is effective, the stress level should be reduced in proportion to E_x and A_x for b = 7.6 inch;

$$\Delta A_s = 7.6 \times .041 \times \frac{25.5}{29.0} = .274 \text{ in}^2$$

$$A_{st} = .3 \text{ in}^2, \; \Delta_{tot} = .3 + .274 = .574 \text{ in}^2$$

$$f_{tx} = 61,296 \times \frac{.3}{.574} = 32,200 \text{ psi @ } E = 29 \times 10^6$$

The actual skin stress is

$$f_{t_{x_{skin}}} = 32,200 \times \frac{25.5}{29} = 28,300 \text{ psi ultimate}$$

Internal Pressure — 10 psi acts at the same time imposing a biaxial tension.

Two conditions need to be considered:

1. Skin carries pressure in hoop tension, or
2. stringer carries pressure to frames.

Condition 1 - Hoop Tension: The box surface is formed with a 113 inch radius.

Chordwise tension

$$f_{ty_p} = p_o \frac{R}{t} = \frac{10 \times 113}{.041}$$

$$= 27,500 \text{ psi ultimate}$$

Longitudinal pressure will be divided between the top and bottom surface.

$$A = 205 \text{ in}^2$$

$$C = 29.65 \text{ in}$$

$$P = 10 \times 205 = 2,050 \text{ lb}$$

Bottom Surface

$$w_p = \frac{2,050}{2 \times 29.65} = 34.5 \text{ lb/in tension}$$

$$f_t = \frac{34.5}{.041} = 840 \text{ psi}$$

$$f_{tx} = 840 + 28,300 = 29,140 \text{ psi ultimate}$$

$$f_{ty} = 27,500 \text{ psi tension}$$

$$\tau_{xy} = -2,720 \text{ psi}$$

The Hewlett Packard is used to resolve these stresses into σ_1, σ_2 and τ_{12} axes.

				Input U.D.		
$f_{1_{max}}$	= 36,332 psi	+45°		E_{11}	=	29×10^6
$f_{2_{max}}$	= 26,472 psi	-45°		E_{22}	=	20×10^6
				ν	=	.29
τ	= $\mp 5,514$ psi	$\pm 45°$		G	=	8.3×10^6
					$(0, \pm 45)_s$	

Condition 2 - Stringer Bending: Tension on Bottom Surface.

Since the spar may not provide hoop tension reaction to the skin, the total pressure load is assumed acting on the stringer as a running load.

$$w = 10 \times 7.65 = 76.5 \text{ lb/in ultimate}$$

The box consists of 5 spars.

The peak moment occurs at the support between the first and second span

$$M = .105 \times 76.5 \times \overline{17.3}^2$$

with tension on the inside (crown of hat) of the box. Tension in the skin occurs at mid span for

$$M_{\mathbb{C}} = .046 \times 76.5 \times 17.3^2 = 1,053 \text{ in/lb}$$

Top Skin   Primary Compression

Max Bending + Stringer Bending

$$\left.\begin{array}{lll} f_{x_b} & = & -43,200 \\ f_{x_{po\,str_b}} & = & -11,500 \\ \end{array}\right\} 54,700 \text{ psi} \qquad \left\{\begin{array}{lll} f_1 & = & -68,307 \\ f_2 & = & 26,265 \\ \tau_{12} & = & -38,299 \end{array}\right.$$

f_y = 27,500 psi Hoop Stress

f_{xy} = 2,500

$$\left.\begin{array}{lll} f_x & = & -54,700 \\ f_y & = & 0 \\ f_{xy} & = & 2,500 \end{array}\right\} \quad \begin{array}{lll} f_1 & = & -65,397 \\ f_2 & = & 23,408 \\ \tau_{12} & = & 24,632 \end{array}$$

Skins buckle between stringers at 11,800 psi for 3.5 in. spacing for .041 skin $\pm 45°$. Reference Figure 2.32.

Summary of Skin Loading

Bottom Skin   Primary Tension

Maximum Bending + Hoop Tension due to Pressure

f_x = 28,300 psi   (skin fully effective)

10 psi Pressure

$$\left.\begin{array}{lll} f_{x_p} & = & \underline{840 \text{ psi}} \\ & & 29,140 \text{ psi} \\ f_y & = & 27,500 \\ \tau_{xy} & = & -2,720 \text{ psi} \end{array}\right\} \quad \begin{array}{lll} f_1 & = & 36,332 \text{ psi} \\ f_2 & = & 26,472 \text{ psi} \\ \tau & = & 5,514 \text{ psi} \end{array}$$

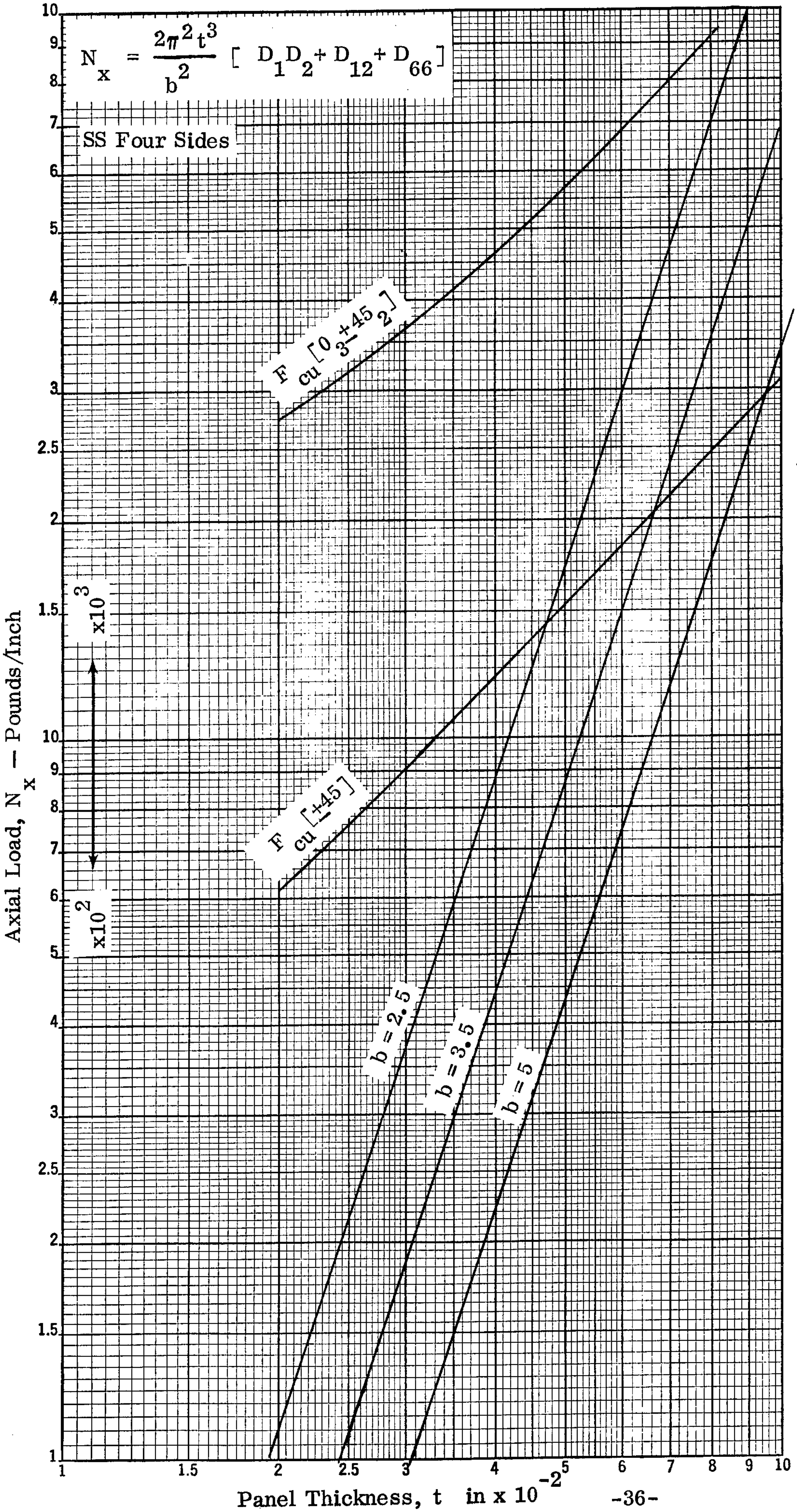
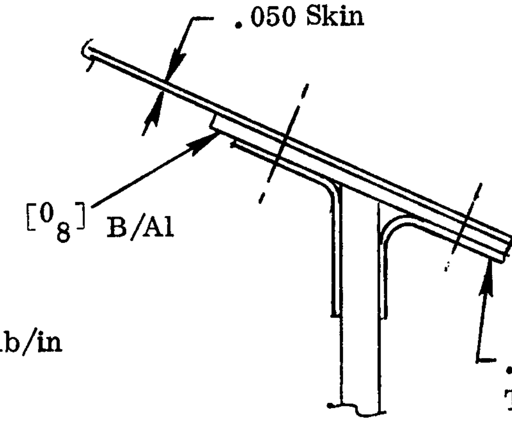


Figure 2.32. P_x, Axial Load Capacity of Crossplied Boron/Aluminum Buckling of Simply Supported Panels A/B = ∞

Maximum Bending + Stringer Bending due to Pressure Skin

$$\left.\begin{aligned} f_x &= 29{,}140 + 5{,}640 = 34{,}780 \text{ psi} \\ f_y &= 0 \\ \tau_{xy} &= -2{,}720 \text{ psi} \end{aligned}\right\} \quad \begin{aligned} f_1 &= 41{,}582 \\ f_2 &= 15{,}796 \\ \tau_{12} &= 15{,}661 \end{aligned}$$

<u>Skin Attachment to Spar</u>



Hoop Tension ~ 10 psi

$R = 113$ inches

Chord Load

$p_t = 113 \times 10 = 1{,}130$ lb/in

Case 2 Torsion

$f_s = 20{,}990$ psi in skin
$t = .041$

$q = 860$ lb/in

$$p_R = \sqrt{\overline{860}^2 + \overline{1130}^2} = 1{,}420 \text{ lb/in}$$

Attachment is by 2 rows of rivets through skin to cap (-11 & -17)

-11 and -17 are bonded together by FM123-2 adhesive

Check NAS 1739B5 and MS20427M Monel Rivets.

B4	B = .225	Nom Dia = .14 x 4 = .76 in
B5	B = .286	= .173 x 4 = .692 in

From MIL-HDBK5B, Table 8.1.4.2.2(h)

Try B5 @ .70 spacing. $p_s = 265 \times 1.5 = 398$ ultimate

MS20427M Monel Rivet 5/32 $\quad 402 \times 1.5 = 603$ ultimate

$$\Sigma P_s = \overline{1001} \text{ ultimate}$$

$$Q = \frac{1001}{.70} = 1{,}430 \text{ lb/in}$$

$$M.S. = \underline{0.0}$$

Stringer Clips — Middle Ribs
72C0178

Each stringer transmits the 10 psi pressure to the frames.

Maximum stringer spacing is on the lower surface at 7.07 inches.

$$p_t = 7.07 \times 17.33 \times 10 = 1{,}220 \text{ lbs}$$

Each stringer has 2 angle clips mounted by NAS1738 rivets, 2 per side.

$$t = .041$$

$$p_{su} = 427 \text{ lb/rivet} \quad \text{MIL-HDBK5b}$$

$$M.S. = \frac{4 \times 427}{1220} - 1 = \underline{+.40}$$

Middle Frames Cap Section

Try 18-8 1/2 Hard

$$F_{cy} = 102 \text{ ksi}$$

For $f_c = 100{,}000$ psi

$$A_{reqd} = \frac{19{,}600}{100{,}000} = .196 \text{ in}^2$$

$$E_s \leq E_{B/Al} = 26 \times 10^6$$

$$A_s = .196 - .041 \times .5 = .170 \text{ in}^2$$

For $\lambda_s = 2.5 \quad t = .170/2.5 = .068$

Buckling of Channel Section

Try .080 1/2H $\quad \lambda = .17/.08 = 2.125$

$\lambda_{chan} = .28\pi = .88 = .8800$ in

$\lambda_{web} = 1.125 - .64 = \underline{.4850}$

$\Delta_b = (2.5 - 1.3650) = 1.135$

$B = \frac{1.135}{2} + (4t = .32) = .888$

$b/t = .568/.08 = 7.1$

$F_{cc} = 29.5$ ksi

$a/t = \frac{.485}{.08} = 6.625$

$F_{cc} = 70.00$

$\frac{b_x}{b_y} = \frac{.568}{.485} = 1.17$

c.f. = 1.03 $\quad$ Reference Figure 14 Lockheed Stress Manual 126

Corners

$R/t = 3.5 \qquad F_{cc} = 51.5$

$$F_{cc} = \frac{51.5 \times .88 \times .08 + 70 \times 1.03 \times .485 \times .08 + 2 \times .08 \times .568 \times 29.5}{.08(.88 + .485 + 2 \times .568)}$$

$= 113,900/2.501 = 45,500$

MCR = 2.509 $\qquad A = .08(2.51) = .20 \text{ in}^2$

$$F_{cc} = 45,500 \times 2.509 = 114 \text{ ksi}$$

$$f_c = 19,600/.20 = 98,000 \text{ psi}$$

$$\text{M.S.} = \frac{114,000}{98,000} - 1 = \underline{+.16}$$

2.4.3.1 Summary of Margin of Safety.

	MS
Steel end fittings	Large
Load introduction fittings, 72Z0196	+3.47
Stringer load introduction	+0.38
Stringer clip attachment to stringer	+0.40
Buckling of rib longitudinal stiffener	+0.16
Access door shear case	+0.24
bending plus pressure	+0.43
attachments	+0.12
Skin pressure	+0.34
Skin attachment to spar caps	0.0
Skin to rib attachment	+0.54
Spar web shear	Large

2.4.4 WING BOX WEIGHT SUMMARY. This section summarizes the weight of the composite section of the wing box during the program.

When the wing box was initially proposed, an estimate was made for the boron/aluminum wing section and a comparable 7075 aluminum section. These weights were 40 lb. for the boron/aluminum to 55 lb. for the aluminum wing for a weight saving of 27%.

At the initiation of the design, another weight estimate was made which is summarized in Table 2-3. These initial weight estimates did not consider load introduction provisions.

Table 2-3. Initial Weight Estimate

Skins	16.28
Ribs	5.22
Stringers	9.28
Spars	5.85
Door landing	0.46
Fasteners	1.50
Adhesive	0.25
	38.84 lb.

At the conclusion of the detail design activity the weights were again calculated. This calculation is summarized in Table 2-4.

Table 2-4. Calculated Wing Weight

Skins	18.31
Ribs	12.8
Stringers	9.40
Spars	12.21
Door landing	1.40
Fasteners	1.63
Adhesives	.60
	56.35 lb.

This total weight of 56.35 lb. compares with weight of 56.98 lb. measured during final assembly.

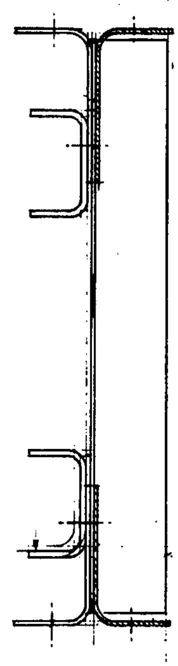
The final calculated weight and the measured weight included some load introduction provisions and are therefore not comparable to the initial weight summaries.

To make a meaningful comparison several adjustments were made to the weights. Since none of the initial estimates included local reinforcement for load introduction, these items were subtracted from the spar and stringer weights.

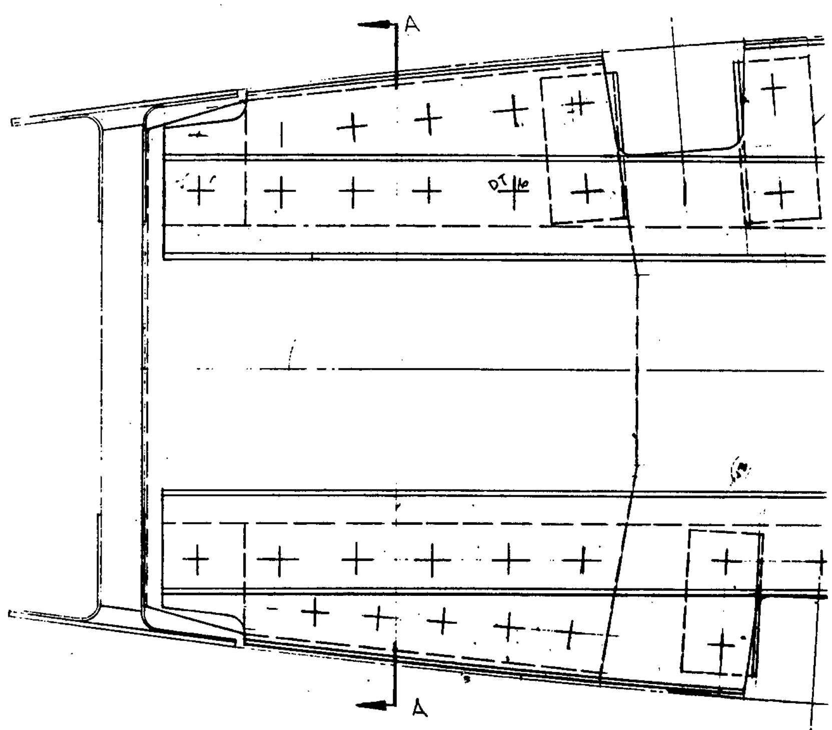
During the final stress analysis the need for chordwise rib braces was determined. Due to budget and schedule limitations, these parts were fabricated from steel although they were intended to be boron/aluminum. See Figure 2.33. The rib weights were then adjusted to the ratio of the density of the two materials for the braces. The final adjusted wing weight for the boron/aluminum wing was calculated to be 43.45 lb. The breakdown is shown in Table 2-5.

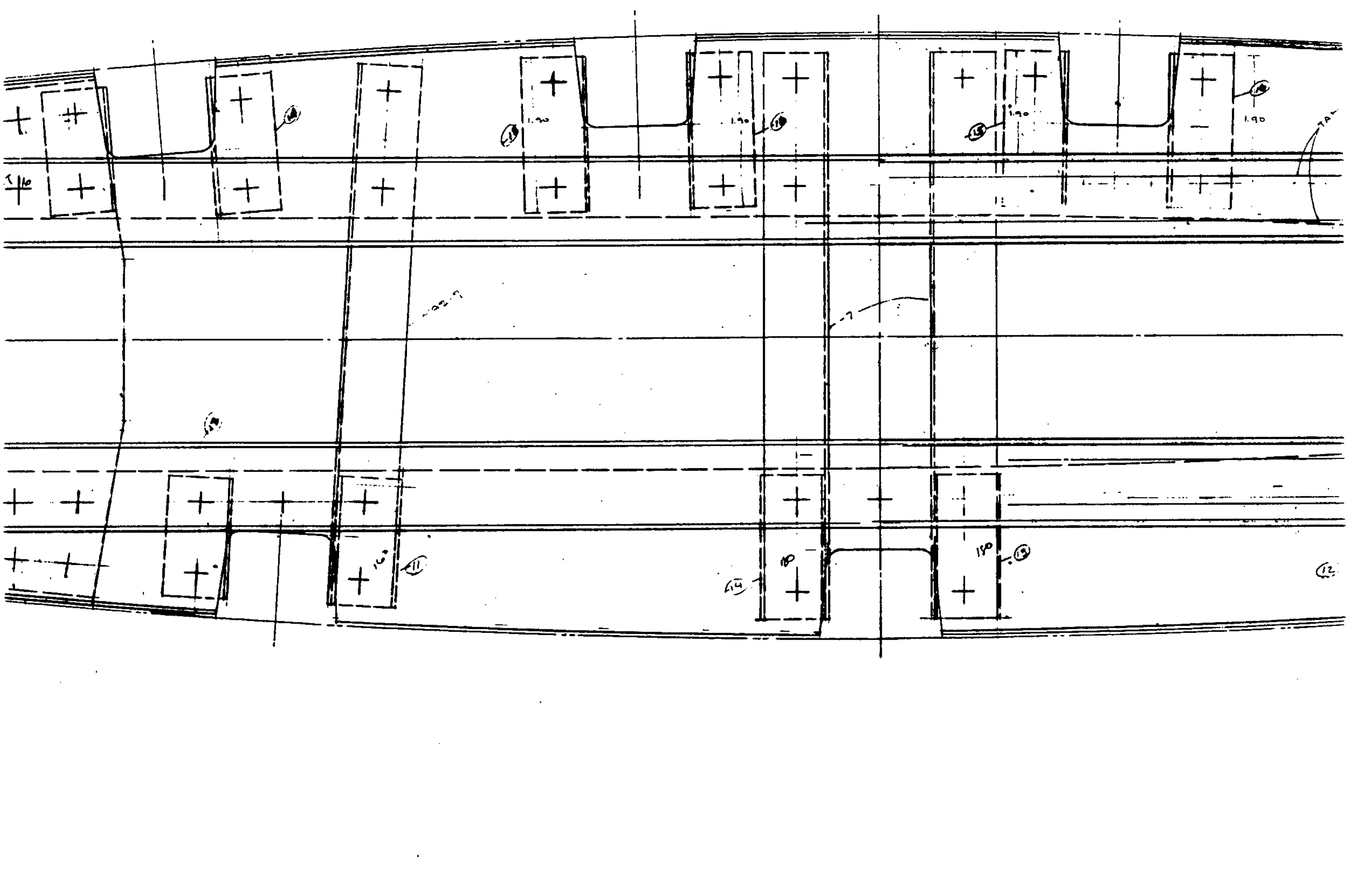
Table 2-5. Adjusted Wing Weight

Skin	18.31
Ribs	8.91
Stringer	7.07
Spars	6.02
Door	1.40
Fasteners	1.46
Adhesive	.28
	43.45 lb.



SECT A-A





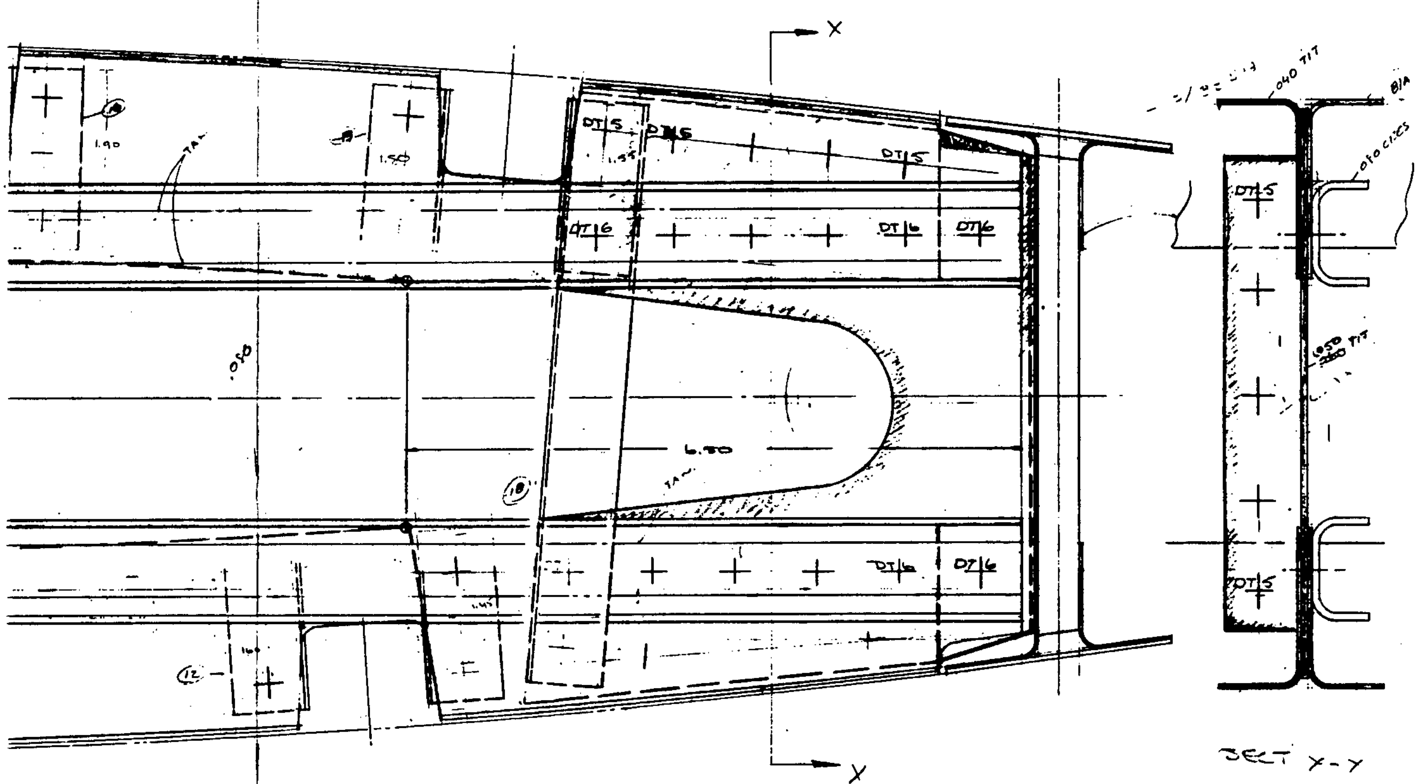


Figure 2.33. Redesigned Center Rib.

③

The aluminum baseline wing weight was then adjusted to account for the change in composite wing length from 48 inches to 52 inches. The adjusted aluminum weight is 57.75 lb. The total weight saving of the boron/aluminum wing was 24.8%.

3

SUBCOMPONENT TESTING

3.1 INTRODUCTION

During the initial design phase, several components of the wing structure were identified for more detailed investigation. To verify their structural integrity, subcomponent test specimens were designed to duplicate these areas.

The subcomponent specimens utilized are summarized below.

Part No.	Function	Number Req'd.
—	Spotwelded specimen	16
72C0172-1	Stringer crippling test, spotwelded	3
72C0172-3	Stringer crippling test, riveted	3
72C0173-1	Composite load introduction, spotwelded	1
72C0173-3	Composite load introduction, riveted	1
72C0174-1	Tension load introduction	1

3.2 SPOTWELD SPECIMENS

Both spotwelding and riveting were initially contemplated for assembly of the stringers for the wing skin panels. In order to evaluate the adequacy of the joints lap shear and cross tension spotweld specimens were fabricated and tested. The materials used were identical to the skin and stringer material planned for the wing; i.e., $[0_6]$ stringers and $[\pm 45/0_3]_s$ skins. The specimens were tested at room temperature and at 300F.

The stress analysis determined that a minimum cross tension strength of 50 lb. per spot would be necessary at a pitch of 0.63 inches. A safety factor of 3 was imposed resulting in a minimum acceptable cross tension strength of 150 lb. per spot.

Table 3-1 shows that this value was exceeded both at 70F and 300F.

The lap shear values were adequate and exceeded the strength of 1/8-inch flush rivets at a similar pitch. The increase in the strength at 300F is characteristic of boron/

Table 3-1. Skin-Stringer Spotweld Test Data

	Load (lb)	Test Temperature (F)
Cross Tension	200	70
	191	
	207	
	200 Avg.	
	207	300
	200	
	218	
	181	
	202 Avg.	
Lap Shear	765	70
	860	
	880	
	740	
	811 Avg.	
	1022	300
	1025	
	1074	
	987	
	1027 Avg.	

aluminum spotwelds; the lap shear strengths usually returning to 70F values around 500F.

From this testing it was concluded that spotwelding was a structurally sound assembly method for the attachment of skin and stringers. For other reasons specified in Section 2 spotwelding was utilized, however.

3.3 CRIPPLING SPECIMENS

The wing skin stiffeners were to be checked for local crippling loads by means of short column crippling specimens. An L'/ρ of 12 has been utilized for all boron/aluminum crippling testing at Convair. The effective column length L' is defined by $L' = L/\sqrt{c}$ where c is the end fixity coefficient. Assuming a fixity of 3.6 for flat ended specimens yields a specimen length of 9.5 inches. Three specimens were made using spotwelds and 3 specimens were assembled using flush blind rivets and an adhesive between the skin and stringer. The ends of each specimen were potted

with Epon 934 aluminum filled epoxy and when cured were machined flat and parallel. Figures 3.1 and 3.2 show a typical spotwelded specimen while Figures 3.3 and 3.4 show a typical riveted specimen.

3.3.1 TESTING OF CRIPPLING SPECIMENS. The testing was accomplished on a 120,000 pound Tinius Olsen universal testing machine at a cross head speed of 0.001 to 0.002 inches per minute.

The test loads are summarized in Table 3-2.

Table 3-2. Summary of Test Loads.

	Specimen	Load
Spotwelded	72C0172-1 #1	22,720
	72C0172-1 #2	23,010
	72C0172-1 #3	22,490
		22,750 Avg.
Riveted & Bonded	72C0172-3 #1	34,050
	72C0172-3 #2	39,300
	72C0172-3 #3	32,350
		31,900 Avg.

Figures 3.5 through 3.8 show the specimens after testing. The spotwelded specimens particularly display classical buckling behavior.

Figure 3.9 shows clearly the buckled hat section with no fractures evident.

3.3.2 TEST ANALYSIS. The strength of each type of crippling specimen was calculated using nondimensional crippling curves (Ref. 2) developed for no edge free and one edge free configurations. The calculated strength for the spotwelded specimens is 22,430 pounds which is compared to the section test strength. The calculated strength to test strength ratio is:

$$R = \frac{22,430}{22,750} = \underline{.99} \text{ for the spotwelded assembles.}$$

The strength calculations for the riveted and bonded flanges modifies the capacity of the flanges.

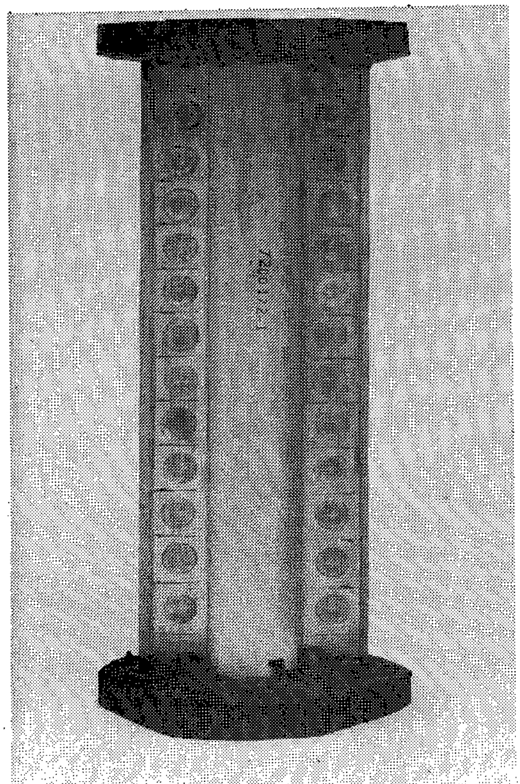


Figure 3.1. Stringer Side of Typical Spotwelded Crippling Specimen. (Neg. 126910B)

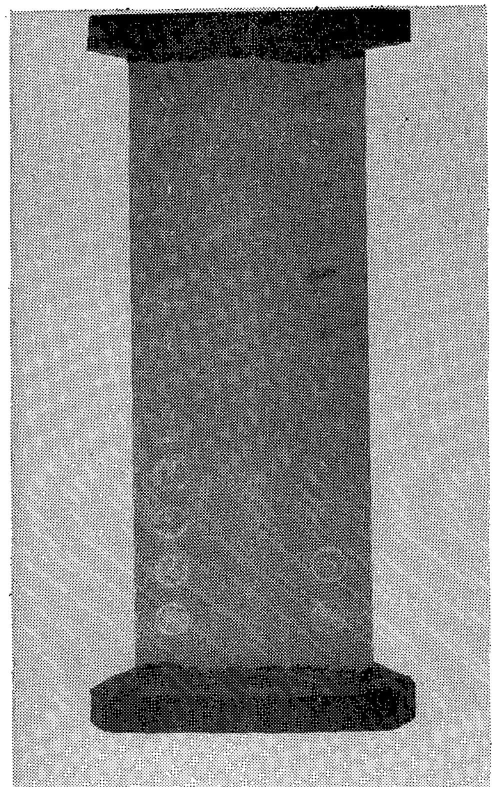


Figure 3.2. Skin Side of Spotwelded Crippling Specimen. (Neg. 126907B)

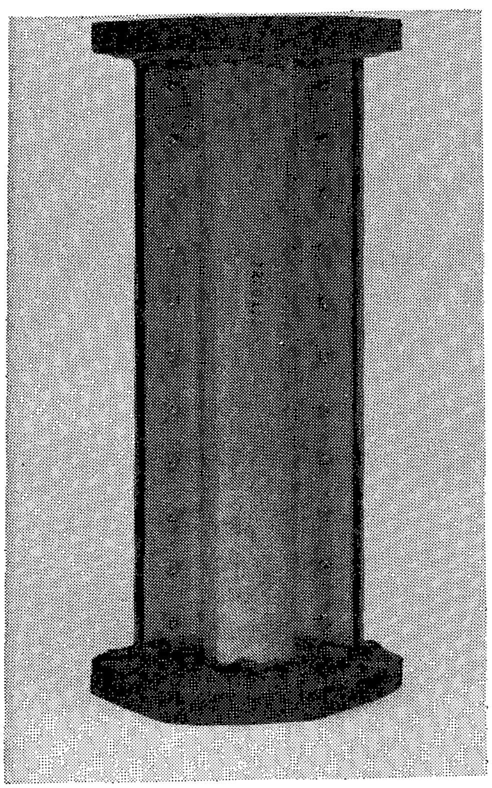


Figure 3.3. Stringer Side of Typical Riveted Crippling Specimen (Neg. 126909B)

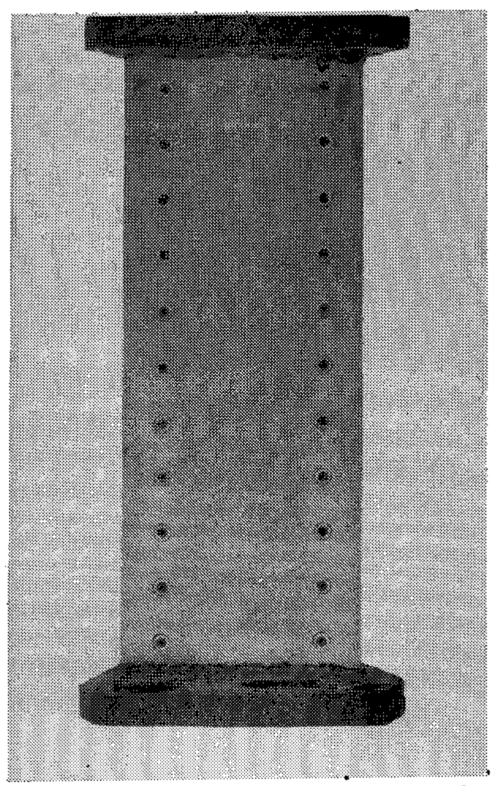


Figure 3.4. Skin Side of Riveted Crippling Specimen (Neg. 126908B)

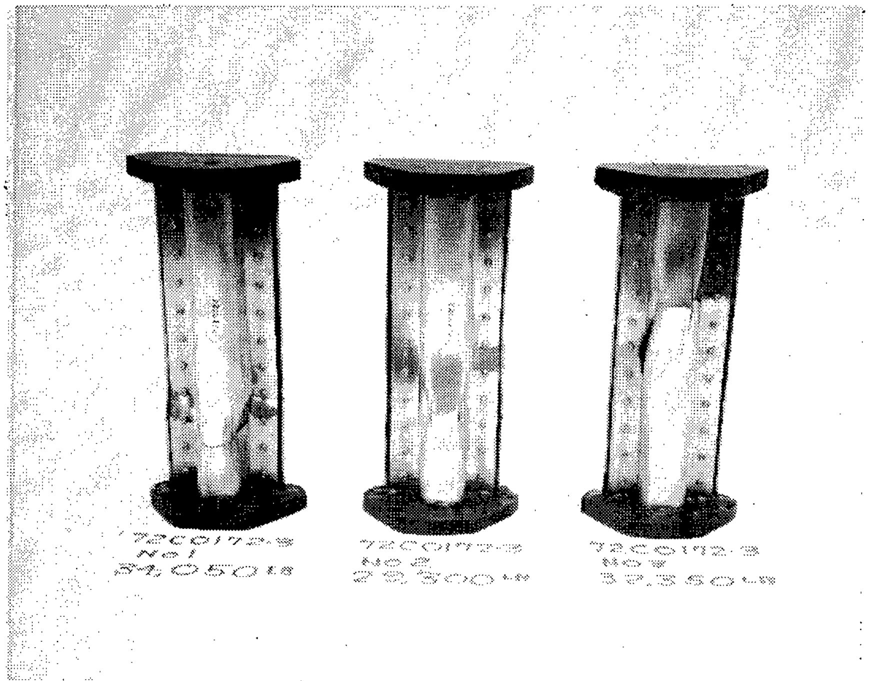


Figure 3.5. Failed Riveted Crippling Specimen Showing Fracture of the Boron Aluminum Stringer (Neg. 126984B)

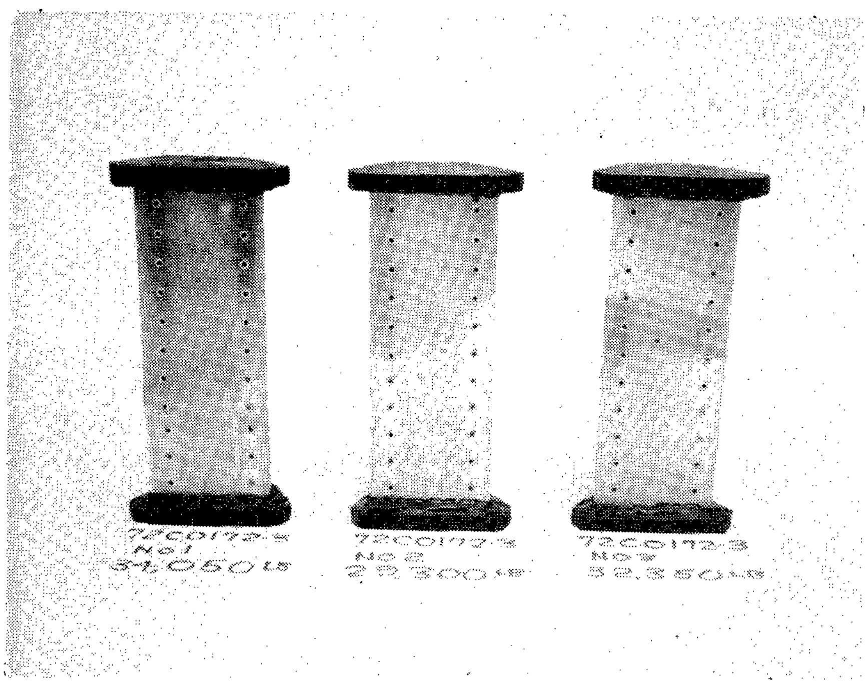


Figure 3.6. Failed Rivet Crippling Specimen Skin Side (Neg. 126986B)

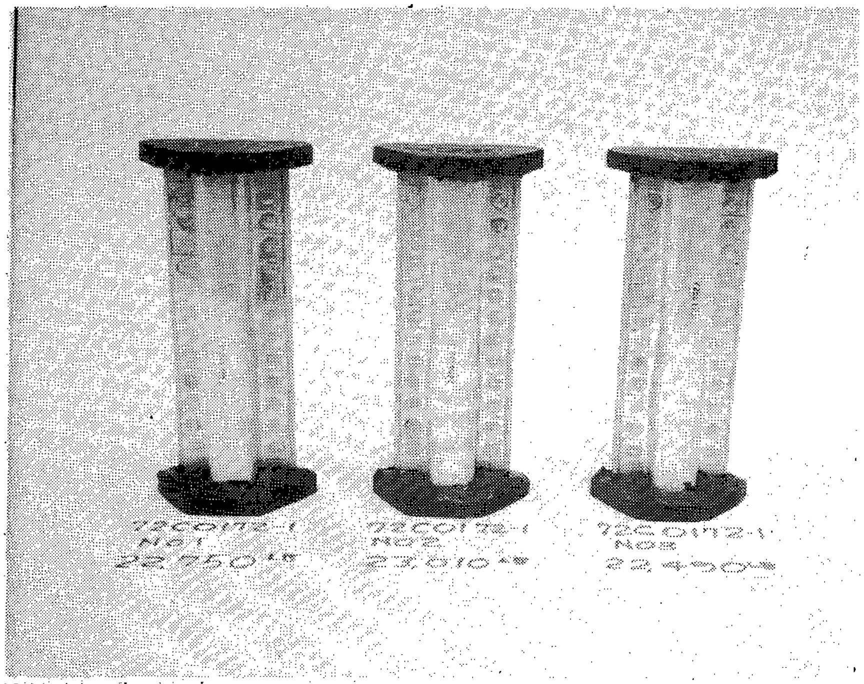


Figure 3.7. Failed Spotwelded Crippling Specimens Showing Classical Buckling of Stringer Elements. (Neg. 126985B)

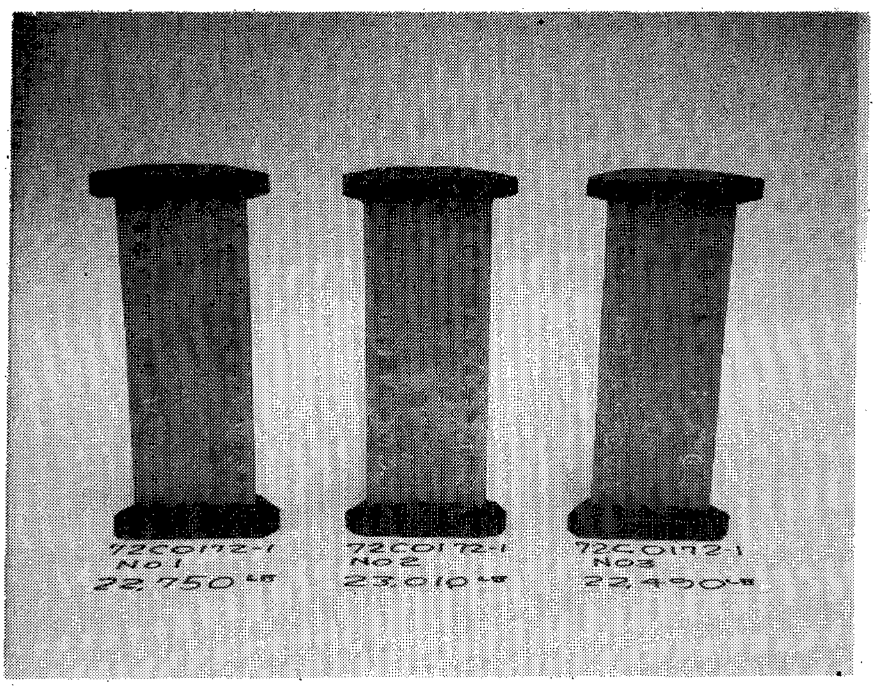


Figure 3.8. Failed Spotwelded Crippling Specimens Skin Side (Neg. 126987B)

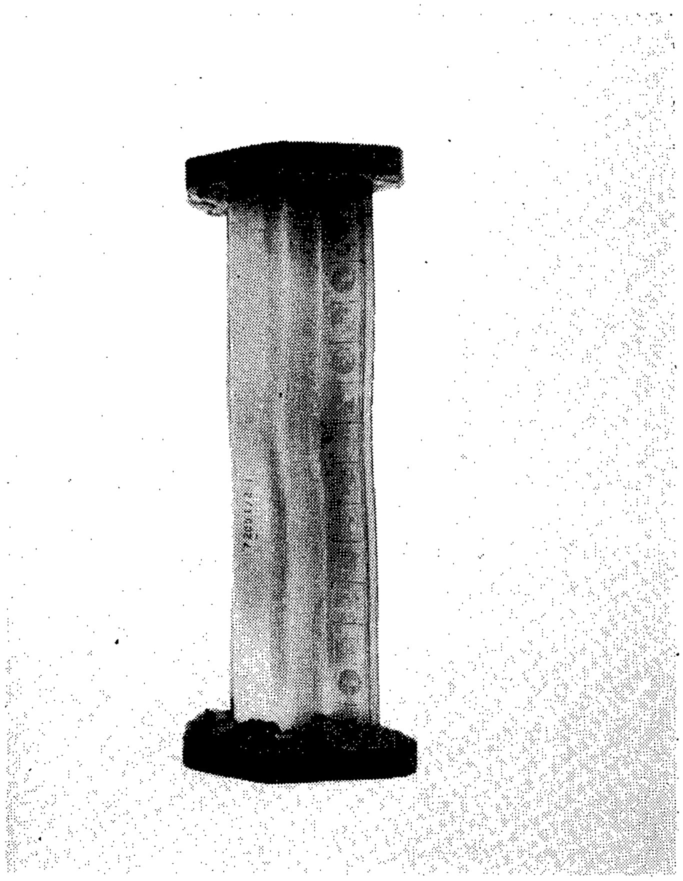


Figure 3. 9. Spotwelded Crippling Specimen Showing Buckles in all Elements of the Hat and Skin. (Neg. 126988B)

P_{all} = 32,000 pounds for a test load average of 31,900 pounds and a calculated strength to test strength ratio:

$$R = \frac{32,000}{31,900} = \underline{1.0+}$$

The riveted assemblies show capacity 31,900 pounds average. The increase can be explained by additional stability resulting from the bond between skin and flange, which greatly increases the buckling load of this element.

3.3.3 CONCLUSIONS.

1. The hat stringer sections demonstrate adequate strength for the wing box design.

2. The analysis reveals an urgent need for improved buckle and crippling information for design allowable predictions, particularly for cross ply boron/aluminum. Both flat plate and R/t data is needed.

3.4 LOAD INTRODUCTION SPECIMENS

Historically the aspect of composite structures which has caused the most problems is that of load introduction. The load introduction from the steel end fittings to the skin and stringer were considered to require substantiation by testing. The final configuration of the stringer end treatment shown in Figure 2.7 was duplicated in three test sepcimens which included a section of composite skin and stringer and a portion of steel end fitting. Two specimens had flat ends suitable for compression testing (Figure 3.10). One of these specimens had the skin and stringer assembled by spotwelding while the other had bonded and riveted construction.

The third specimen was equipped with end fittings to permit tension loads to be applied. See Figures 3.11 and 3.12. The tension specimen had a wider skin section than that provided on the compression specimens. This was to permit evaluation of tension load transfer from the steel end fitting into the composite skin of the wing.

3.4.1 TESTING. The specimens were tested on a 120,000 pound Tinius Olsen universal testing machine at a loading rate of 0.001 to 0.002 inches per minute. Both the compression and tension specimens were provided with a lateral support at the junction of the composite and steel end fitting. The lateral support duplicated the support provided by the end ribs and was necessary to prevent bending of the specimen due to the eccentricity of the centroid of the splice plate and end plate.

The support consisted of a steel angle clamped to the machine columns supporting a steel plate bearing against the specimen at the proper height.

The compression specimens failed at 21,640 pounds for the riveted specimens (Figures 3.13 and 3.14) and 17,690 pounds for the spotwelded specimens (Figures 3.15 and 3.16). Failure on both specimens consisted of crippling of the skin and stringer away from the loads introduction area. The maximum required compression load was 13,880 pounds. Both of the specimens easily surpassed the required loading.

The tension specimen was also provided with a lateral support at the end rib location to duplicate the conditions in the actual structure. The loads were applied by clevises with bolts through the holes provided at each end. Failure of this specimen occurred at a load of 22,860 lb. The required tension load being 18,390 lb. The specimen therefore demonstrated more than adequate strength.

The failure occurred at the end of the stringer load introduction fitting (Figure 3.17). It is obvious that more care is needed to be taken in the design of the test joint. A better test could have been achieved by using a double joint typical of the composite to steel end of the specimen.

3.4.2 CONCLUSIONS.

1. The design of the skin-stringer to end fitting load introduction was adequately substantiated.

2. Special care is necessary in the design of any load introduction fittings to ensure that all load paths are considered.

Figure 3.10. Stringer Side of Compression Splice Specimens Showing Riveted and Spotwelded Construction.

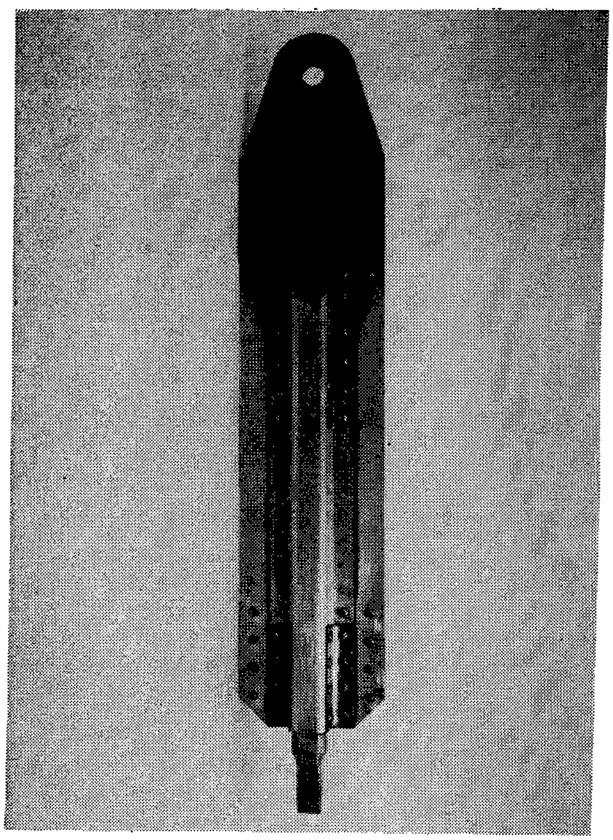


Figure 3.11. Stringer Side of Tension Stringer Splice.

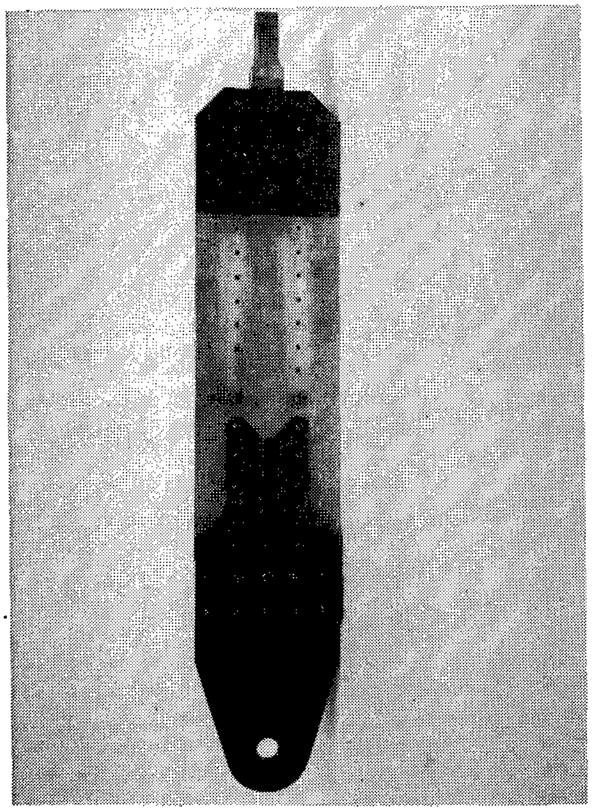


Figure 3.12. Skin Side of Tension Stringer Splice.

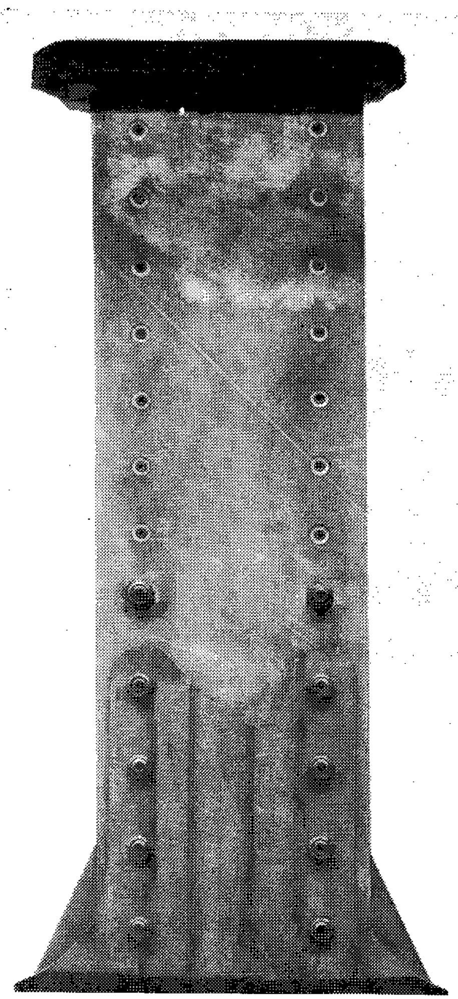


Figure 3.13. Riveted Compression Load Introduction Fitting after Testing.

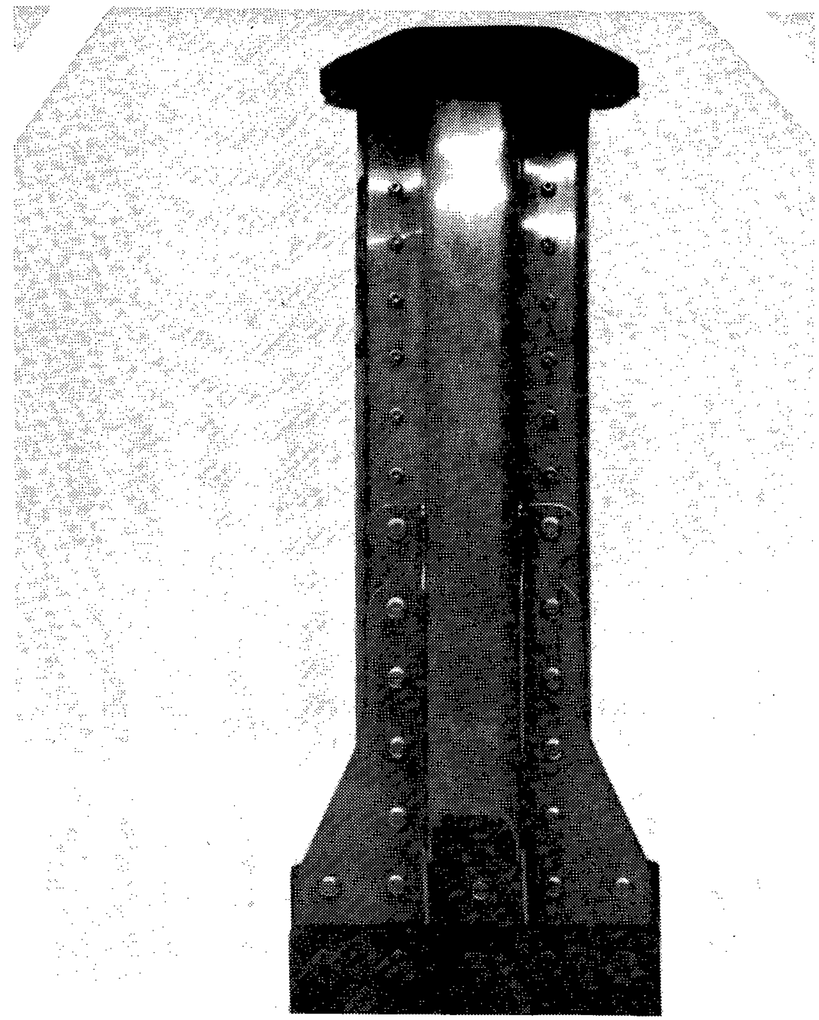


Figure 3.14. Riveted Compression Load Introduction Fitting. The buckle is apparent near the top of the hat just below the end potting.

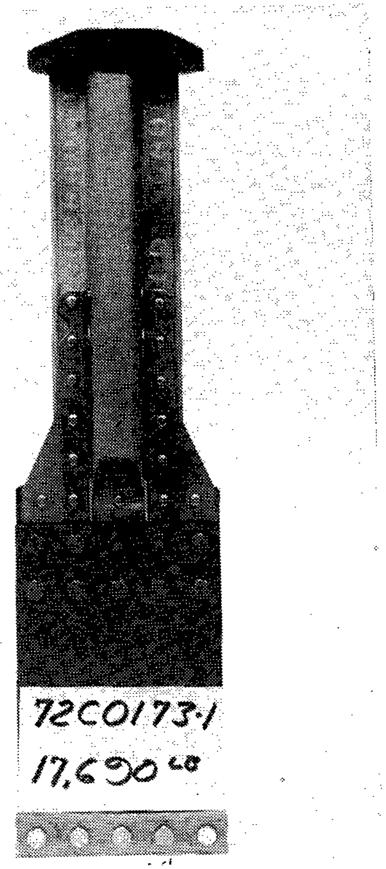


Figure 3.15. Spotwelded Compression Load Introduction Fitting after Testing.

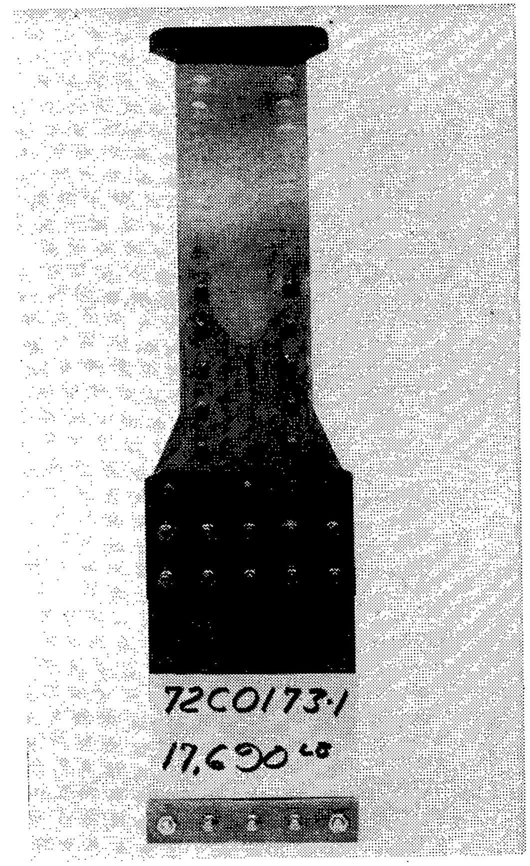


Figure 3.16. Stringer Side of Failed Spotwelded Compression Fitting.

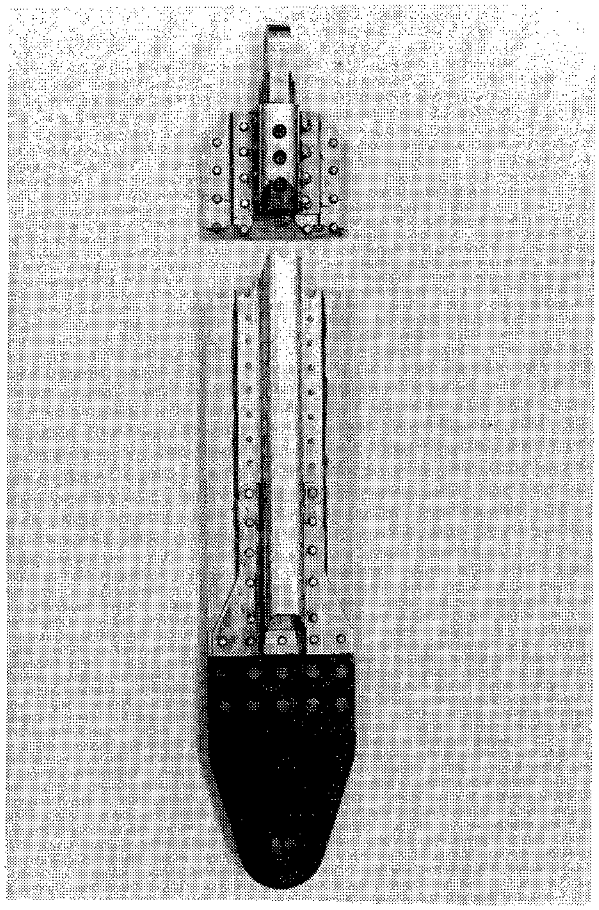


Figure 3.17. Failed Stringer Joint Specimen.

4

FABRICATION

4.1 INTRODUCTION

The principal task of this program was the fabrication of the wing box test specimen and the various subelement test specimens. Convair Aerospace has been developing many advanced fabrication techniques for boron/aluminum during the past several years. Most of these techniques are based upon sheet metal technology which has been used by the aerospace industry for years. It is believed that this approach provides the lowest cost method of applying the material to aircraft structure. These techniques were utilized during the fabrication of the wing box.

4.2 MATERIAL PROCUREMENT AND NDE

All boron/aluminum material for the program was ordered to specification 72C0191 (Appendix A) and was supplied by Amercom, Inc. of Northridge, California. Incoming boron/aluminum was visually inspected and subject to ultrasonic C-scan examination. The material was generally of good to excellent quality with surface finish approaching that of conventional aluminum. One sheet of skin material had a minor blemish on the surface. This was examined by C-scan and radiography and was found to be purely superficial, affecting only 0.0002 to 0.0004 of surface aluminum. Material trimmed from each sheet of incoming composite was made into tensile specimens. Tests of these specimens verified that all material met or exceeded the specification requirements.

4.3 FABRICATION METHODS

The processes utilized are summarized as follows:

a. Shearing

b. Hole punching

c. Diamond sawing

d. Ultrasonic drilling

e. Hot forming

f. Creep forming

g. Diamond routing

h. Dye pnentrant inspection

i. Spotwelding

j. Riveting

4.3.1 The thinner gages of boron aluminum (up to 0.040) can be sheared using conventional metal working equipment when the resulting edge finish is acceptable. Generally the thicker the material the more severe the burr which is left. Many of the smaller, thinner gage parts such as the rib braces and the notches in the ribs were sheared.

4.3.2 HOLE PUNCHING. The punching process has proved a satisfactory and economical approach to producing holes in boron/aluminum sheet material. The inexpensive hand punch shown in Figure 4.1 was used extensively at an equipment cost of only a few dollars. Hand punches do an excellent job up to thicknesses of 0.08 inches. As many as 500 holes can be punched with one set of inexpensive dies. Heavier gages up to 0.10 inch have been punched using more powerful punches. Holes up to 1.5 inches have been made using these processes. Generally holes for fasteners made by punching are cleaned up to final size by reaming with a conventional high speed steel drill or a diamond coated drill.

4.3.3 DIAMOND SAWING. All the boron/aluminum composite material used in the wing box was purchased in large sheets and plates of the proper thickness, and subsequently cut to size. To accomplish this machining task, a large, gantry type unit, designed and built under a previous company sponsored program, was used. The saw shown in Figure 4.2 has an overarm carriage unit with an overall length of 12 feet. A steel-framed table supports the 2 x 12 foot long aluminum table top. The overarm rail contains a rack and pinion drive for moving the carriage containing the diamond blade. The 10 inch diameter by .050 inch thick diamond impregnated cutoff blade is driven by a 2 hp motor at speeds up to 3450 rpm. This basic machining process is rapid, low in cost and is capable of yielding a finished edge surface to accurate dimensions so that assembled structures do not require secondary finishing steps.

4.3.4 ROTARY ULTRASONIC DRILLING. One of the better hole making methods is by use of the rotary ultrasonic machine shown in Figure 4.3. The machine superimposes an ultrasonic impulse on a rotating diamond drill to eliminate the usual "loading-up" of aluminum matrix material on the drill surface. This permits the cutting edges to be exposed to the work material at all times. This tool is particularly useful in hole making in composites too thick to punch and composites in conjunction with conventional materials.

4.3.5 HOT FORMING. The equipment used to form the hat section stiffeners and other sections was a specially designed machine fabricated by Convair Aerospace to hot form boron/aluminum composites. The machine is a 4-post design with 72 inch x 10 inch press platens. The press has 17 inches of daylight and a 4 inch stroke and

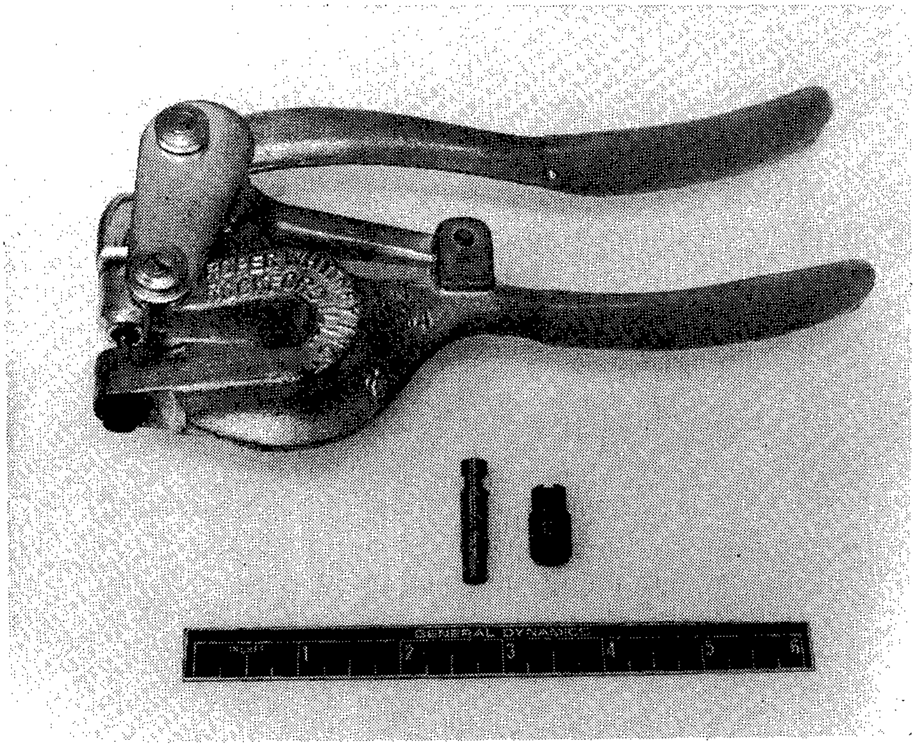


Figure 4.1. Hand Punch with Replaceable Punch Dieset.

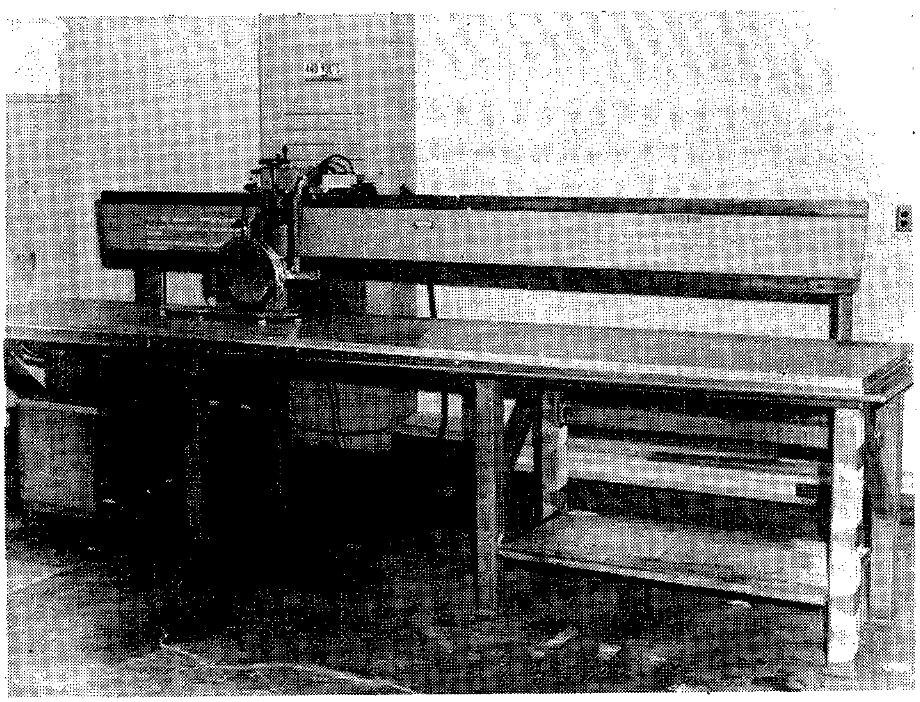


Figure 4.2. Diamond Abrasive Cutoff Machine Used to Size and Trim B/A1 Sheet and Plate Material.

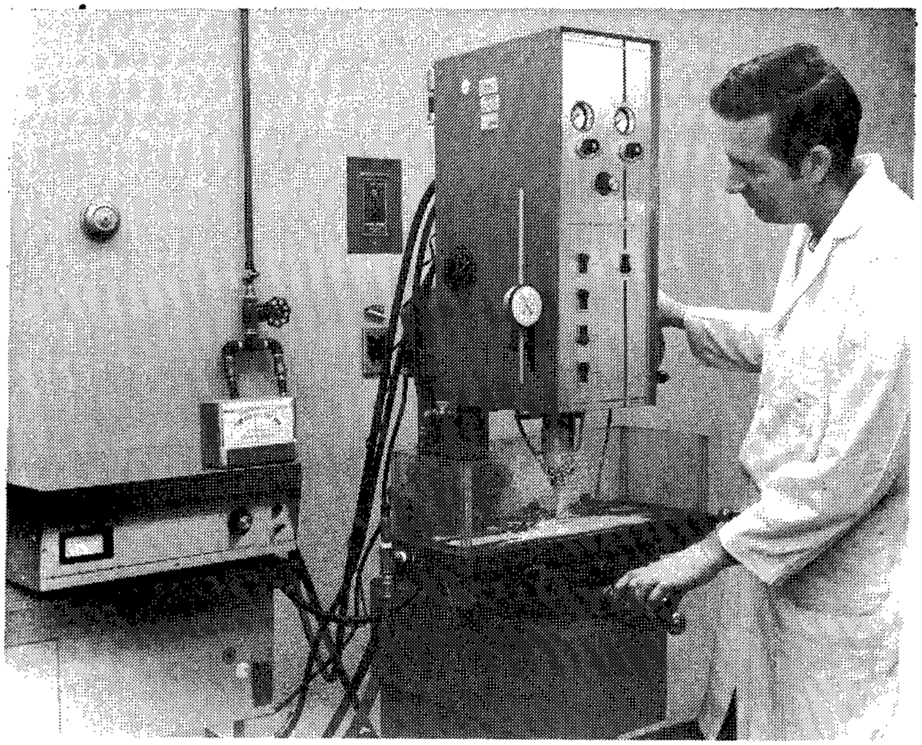


Figure 4.3. Rotary Ultrasonic Machine.

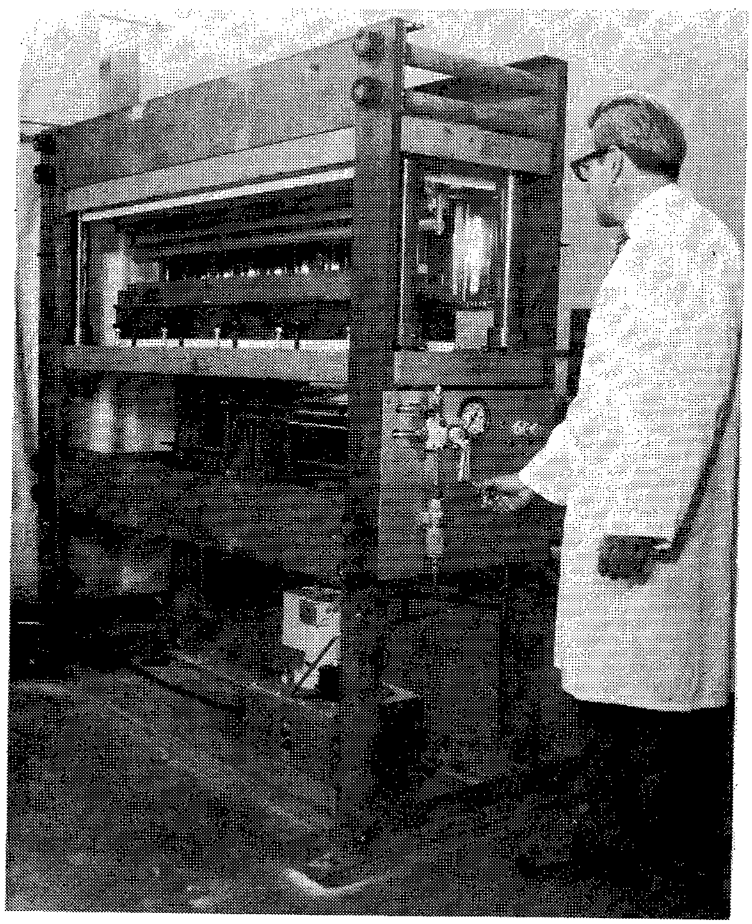


Figure 4.4. Heated Die Forming Press with Sides Removed. (Neg. 122017B)

has a capacity of 35 tons. Figure 4.4 is an overall view of the press. It is designed with the lower platen as a movable platen in order to minimize heat losses (chimney effect) during separation of the tools for loading and unloading. In addition, an important feature of this design is the fine degree of control it offers in the low range of forming loads by eliminating the dead weight effect of the upper platen and tools. The forming rate is infintely variable from 0 to 1 inch/second. The heating chamber of the press is 12 inch square x 60 inch long. It consists of two banks of T-3 radiant lamps mounted on each side of the press to form an oven-like atmosphere for the tools and blank. This effectively provides an isothermal forming operation. Each lamp bank consists of ten reflector assembly modules positioned so that the lamps are mounted vertically. Temperature uniformity along the length of the heating chamber is obtained by varying the lamp density along the length (higher lamp density at the ends of the chamber to offset "end effects"). The reflector surfaces of the modules are gold plated for maximum reflectivity. The modules utilize 1000-watt, 240-volt T3 quartz lamps, and are designed to provide local air cooling of the lamp and seals for longer lamp life. The tooling for forming parts was 90° matched dies fabricated from 316 corrosion resistant steel. The tooling consists of a punch holder, double ended punch, gooseneck punch, Vee die, die holder, and die riser. The tooling arrangements provides for forming the composite material at the center of the heating zone. The system was designed so that the punch is self-centering and both the punch and the Vee die can be changed while hot. See Figure 4.5. Blank indexing is accomplished by locating slots in the blank (which are on the bend centerline) with dowel pins on the centerline of the tools. The blanks are loaded and unloaded from one end of the press.

Based on previous forming experience, the following forming conditions were used:

Temperature	900F
Preheat time	5 min.
Forming time	1 min.
Dwell time	0.7 min.
Backup support	0.025 inch annealed stainless steel

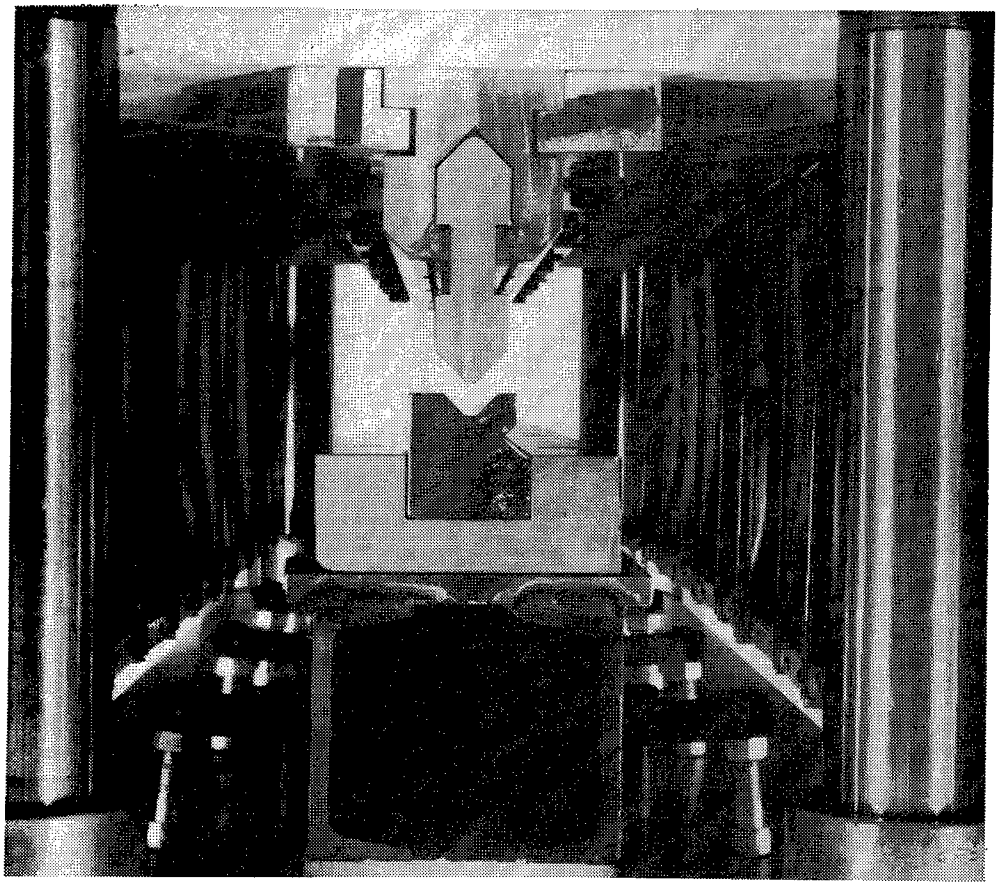


Figure 4.5. Closeup View of Die Holder and Dies in the Hot Forming Press.

The temperature variation from one end of the punch to the other was within 30F; the variation from one end of the Vee die to the other was within 50F. Throughout the heating and forming cycles, the average temperature of the punch was approximately 60F higher than the average temperature of the die. In order to avoid bowing and distortion of the tools, heatup was accomplished in steps at a slow rate. Typically, the tools were brought up to 900F in 3 hours.

For fabrication of hat sections three bends were made with the double ended punch and the fourth bend was made with the gooseneck punch (which was required to accommodate the narrow width of the hat). The double ended punch was used for the three bends because of the better forming action it provides over the gooseneck punch in matching die forming. The double ended punch was designed so that approximately 1/2-inch of contact is made between the punch and die. This design results in a flat surface beyond the radius and minimizes transverse crowning.

Checking the angularity after the first three bends were made indicated that spring-back of approximately 2° was occurring. (The angles had an included angle of 92°.) This was primarily due to the use of the stainless steel backup strip. Prior to forming the fourth bend with the gooseneck punch, the three bends were sized without a backup strip. This brought the angles to 90°. Everlube T-50 was used as a lubricant on both the boron/aluminum blank and the stainless steel backup strip. Visual examination indicated no cracking occurred in any of the bend lines. Straightness of the parts was excellent and the hats could be held absolutely flat with very light finger pressure. The hats were cut and trimmed to length to use in the hat compression, splice specimens, and the wing box.

4.3.6 CREEP FORMING. Creep forming is utilized to form sections to a curved configuration. The boron/aluminum rib caps were creep formed to the contour (113 inch radius) from boron/aluminum angles which had previously been hot formed from sheet (Figure 2.6.)

4.3.7 DIAMOND ROUTING. Irregular shapes such as the rib webs were routed from sheet boron/aluminum using a diamond tool on a conventional tracer operated milling machine.

4.3.8 DYE PENETRANT INSPECTION. All hot formed parts were inspected after forming by conventional fluorescent dye penetrant inspection procedures. All parts used in the wing box were found to be entirely free of cracks as indicated by this process. Prior to inspection, the parts were cleaned by a vapor degreasing.

4.3.9 RESISTANCE SPOTWELDING. A schedule was developed for resistance spotwelding six ply (0.042 inch) unidirectional boron/aluminum sheet. The weld schedule shown in the following table were approximately 0.30 inch in diameter with a single spot lap shear lap strength of 800 to 100 pounds (see Section 3).

Weld Schedule

Machine:	100 KVA
Electrodes:	Class I, 5/8-inch full face, 8-inch tip radius

Heat Input:	8 impluses preheat at 36° phase shift 27 impulses weld heat at 48° phase shift
Forge Delay:	12.5 cycles after start of weld
Pressure:	1050 lbs-weld, 1950 lbs-forge pressure

Several subelement specimens were made using this schedule, however, a decision was made not to use spotwelding in the wing box itself.

4.3.10 RIVETING. Two kinds of rivet type fasteners were used in the assembly of the wing box. The primary fastener was the bulbed Cherrylock fastener, Universal head (NAS 1738) and flush head (NAS 1739) were used. These fasteners were used because they have a controlled expansion, have shear and tensile strengths similar to driven rivets and they also have a large bulbed formed head to eliminate pull-through. Previously NAS 1398 and NAS 1399 rivets were utilized for boron/aluminum, however, these rivets had a tendency to swell the holes in the composite leaving a loose rivet. The bulbed type have overcome this problem.

Steel lock bolts with aluminum collars were used in the highly loaded areas. Hole sizes for these fasteners were generally 0.0005 inch larger than conventionally specified.

All fasteners installed in the wing box were dipped in wet epoxy primer for sealing and corrosion prevention.

4.4 ASSEMBLY

The assembly fixture with the skin and stringers in place are shown in Figure 4.6. Notice the low cost wooden type forms used to give the skin curvature. After the hats were punched on they were used as hole templates for the skins. It was important that the skin be made to assume the natural curved position before the fasteners were attached to minimize stresses. A sufficient number of holes were manually drilled on the fixture so that temporary fasteners (cleco) could be put in place. Figure 4.7 shows the cleco fasteners in place thus allowing the top supports to be pulled back. The remainder of the holes were then punched by removing the skin assembly from the fixture and placing it on a special punch shown in Figure 4.8. The fixture consisted of a C-frame adapted from an old riveting machine to which a screw driven punch had been attached. Notice the automobile steering wheel used by the operator to rotate the screw. The important feature of the machine is the screw driven punch allowing a slow and controlled shearing action on the composite. Rapidly driven punches tend to take more power and make poor holes. Another important feature of the machine is that one man can operate it unaided. This is possible because of the controlled action of the screw and the ease by which the wheel is handled. All the fastener holes

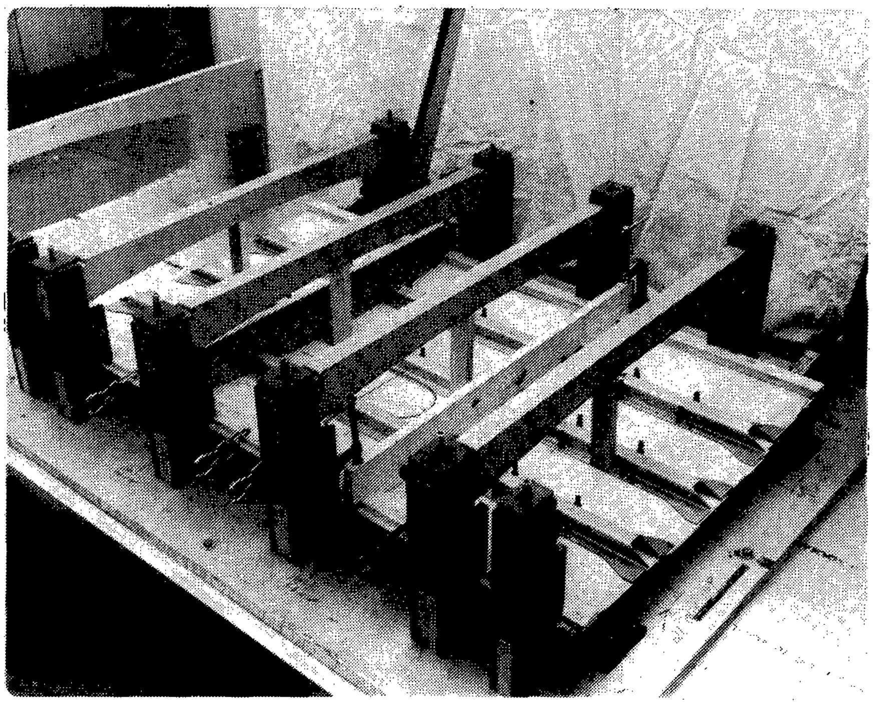


Figure 4.6. Assembly Fixture used to Attach Stringers to Skin.

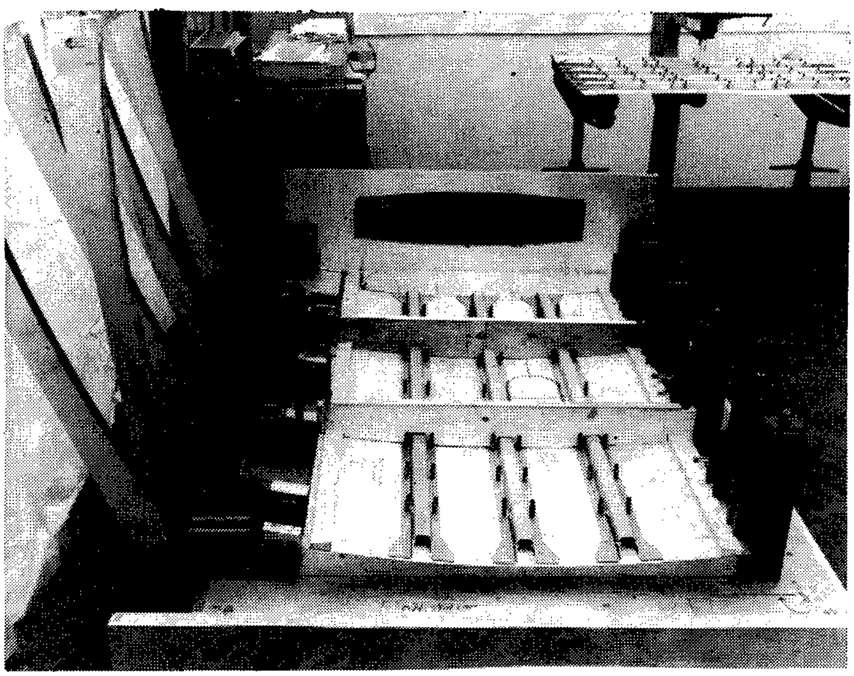


Figure 4. 7. Assembly Fixture with Top Supports Pulled Back Showing Cleco Fasterners in Place.

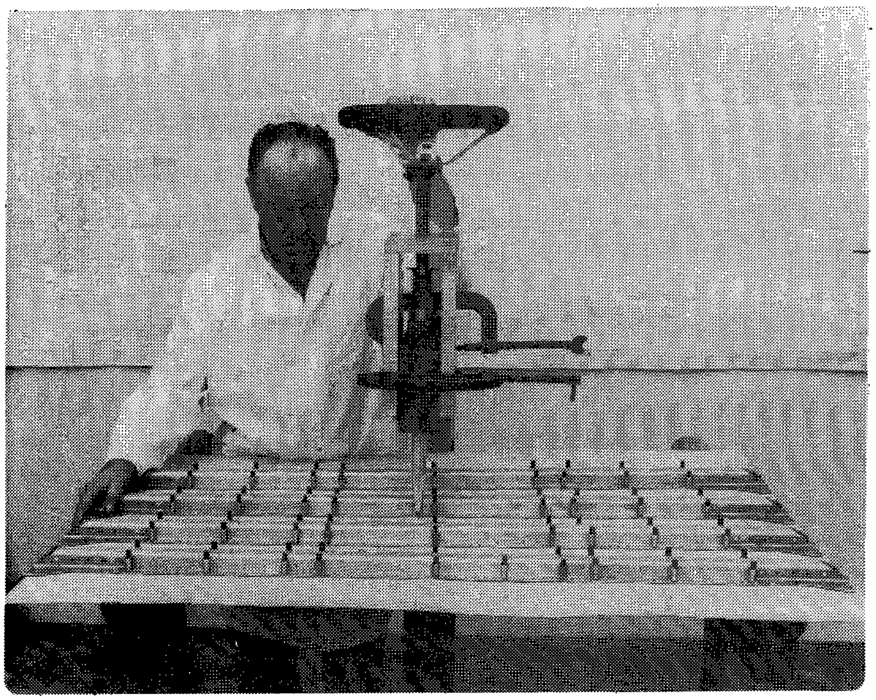


Figure 4.8. Special Punch Used to Put Holes in Skin.

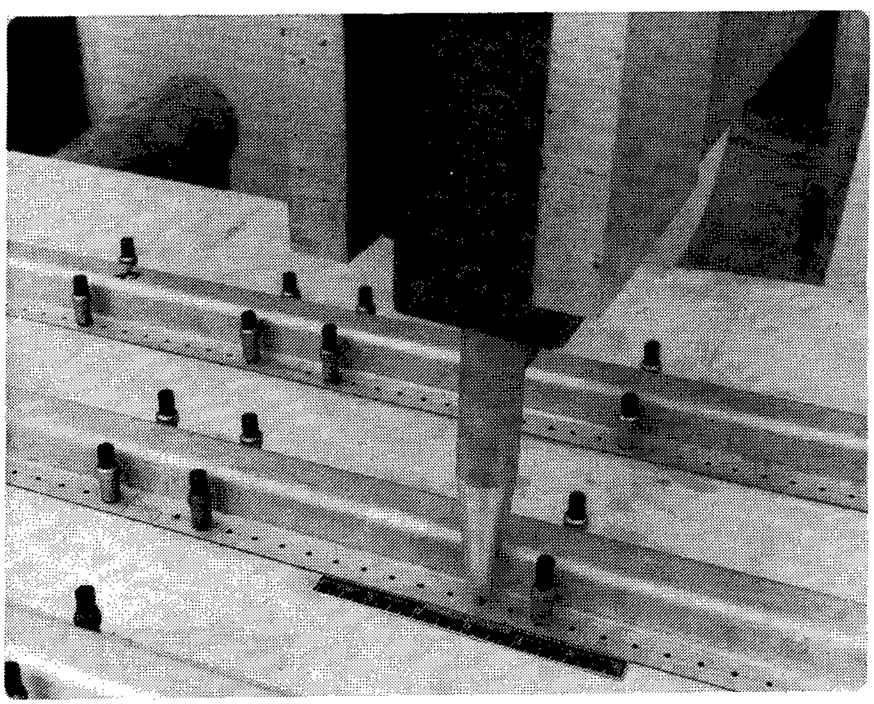


Figure 4.9. Closeup View Showing Punch Transferring the Hole pattern from the Stringer to the Skin.

in the skin were punched in this manner using the punched holes in the hats as templates. A closeup view of the punch is shown in Figure 4.9. The fabrication methods used on boron/aluminum are rapidly approaching those used on ordinary sheet metal. After all the holes had been punched in this manner, they opened to size by a high speed steel drill prior to the final installation of the rivets.

The partially completed spar assembly is shown in Figure 4.10. It consists of an aluminum honeycomb core enclosed between boron/aluminum skins to form the web. Also shown are the completed titanium angles and boron/aluminum caps. Inserts to fill the spaces at the top and bottom of the cores are made from boron/aluminum machined strips. Diamond plated milling cutters made to order and having the proper radius were used to machine the inserts. After all the necessary fastener holes were predrilled and punched, the angles, caps, fillers, web skins, and honeycomb were adhesively bonded together to form completed spar.

The next step in the construction of the wing box was to assemble the steel end fittings and place them into the assembly fixture. The composite spars were then clamped into the fixture and the ribs assembled to the contour blocks. The stringers were removed from the skin panels and fitted into the fixture. When all the detail parts had been fitted, and pilot holes were punched or drilled. The entire composite section was disassembled and each detail marked with its location. All detail parts were cleaned, primed with epoxy primer, and enameled with white acrylic enamel. The detail parts were then again placed into the fixture and fasteners were used to permanently assemble the parts.

Figure 4.11 shows the wing box assembly prior to the final fitting of the skins.

The skins were then again fitted to the structure and the holes brought to finished size. During final installation of the skins, sealant was placed along the spar caps and epoxy adhesive placed along the stringers. Each skin in turn was placed onto the subassembly held in place with clecos and the rivets installed. Figure 4.12 shows the placing of the fasteners in the skin.

Figure 4.13 shows the interior of the wing box with the end rib web not installed. The center ribs are visible beyond.

The completed wing box with strain gages installed is shown in Figure 4.14. The end plates had not been installed at this stage.

Figure 4.15 shows an interior view of the specimen showing the strain gage wiring passing through a yet uninstalled rib web and through a pressure fitting to the outside.

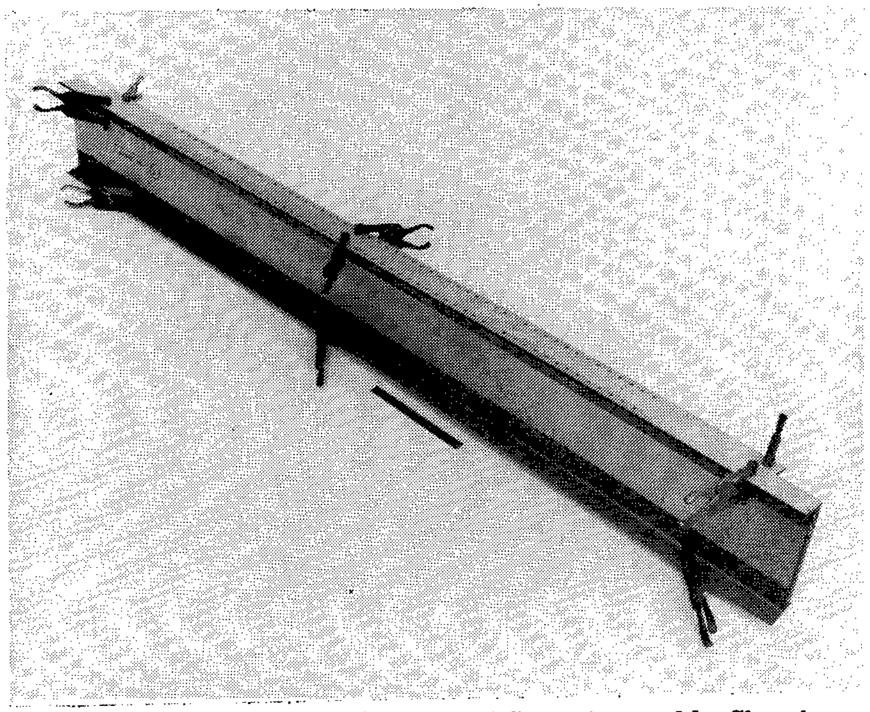


Figure 4.10. Partially Completed Spar Assembly Showing the Angles, Caps, Fillers, Web Skins and Honeycomb Core Clamped together ready for Bonding.

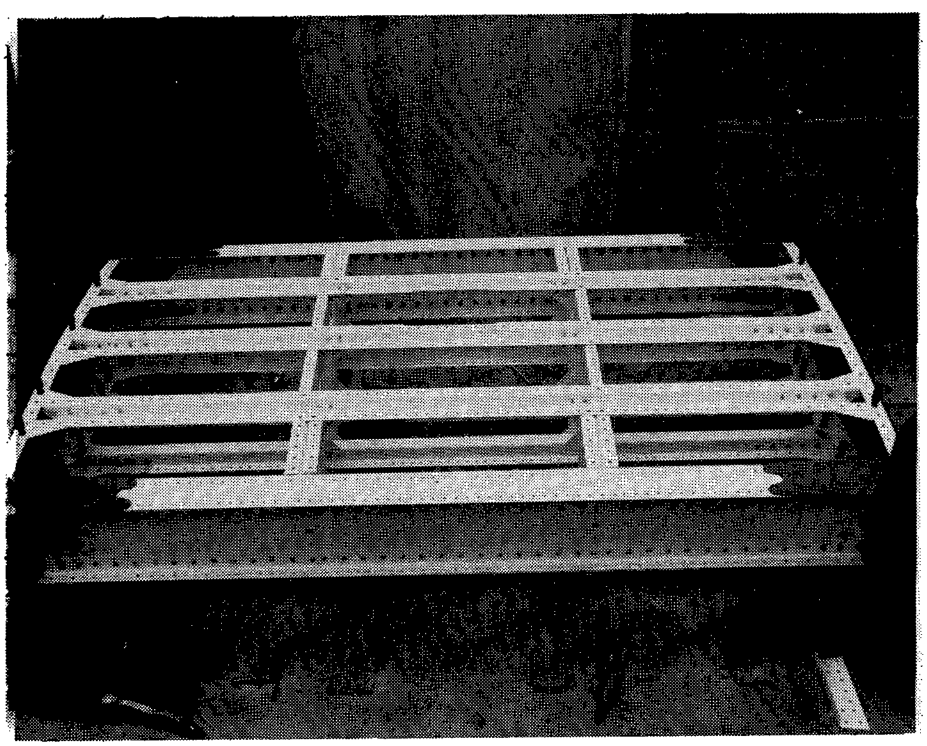


Figure 4.11. Wing Box Assembly Prior to Fitting of the Skin.

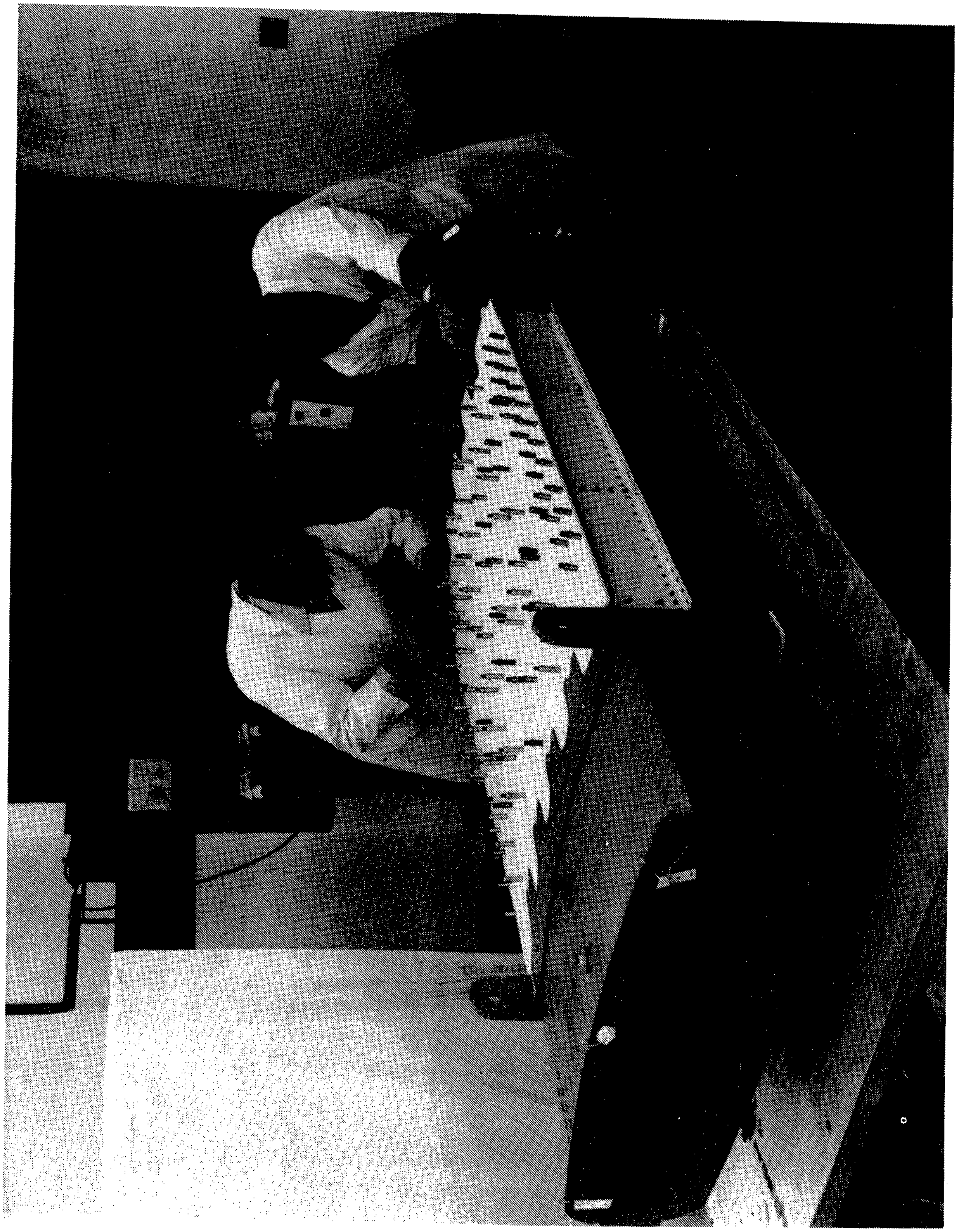


Figure 4.12. Final Assembly of Wing Box Installing Skin Fasteners.

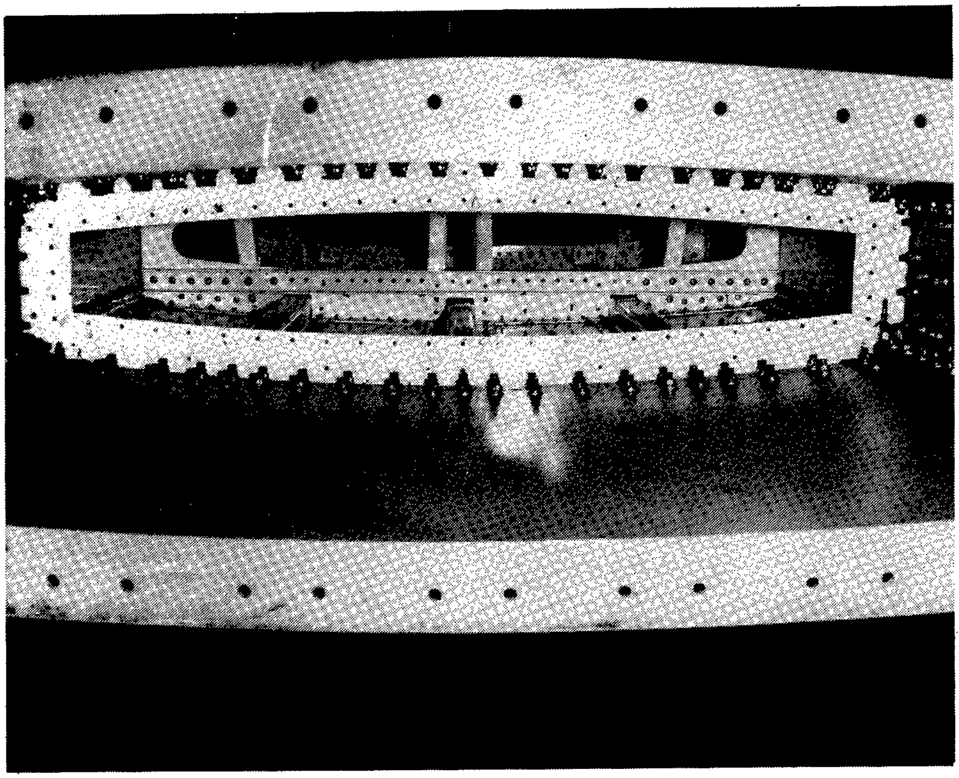


Figure 4.13. Interior View of Wing Box.

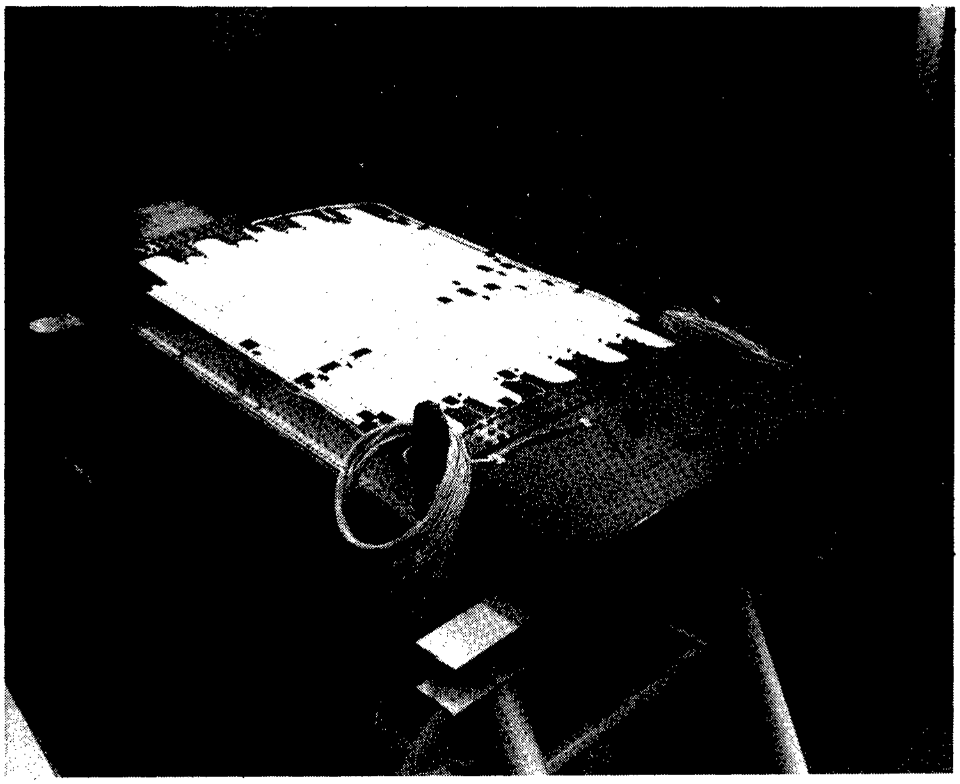


Figure 4.14. Completed Wing Box.

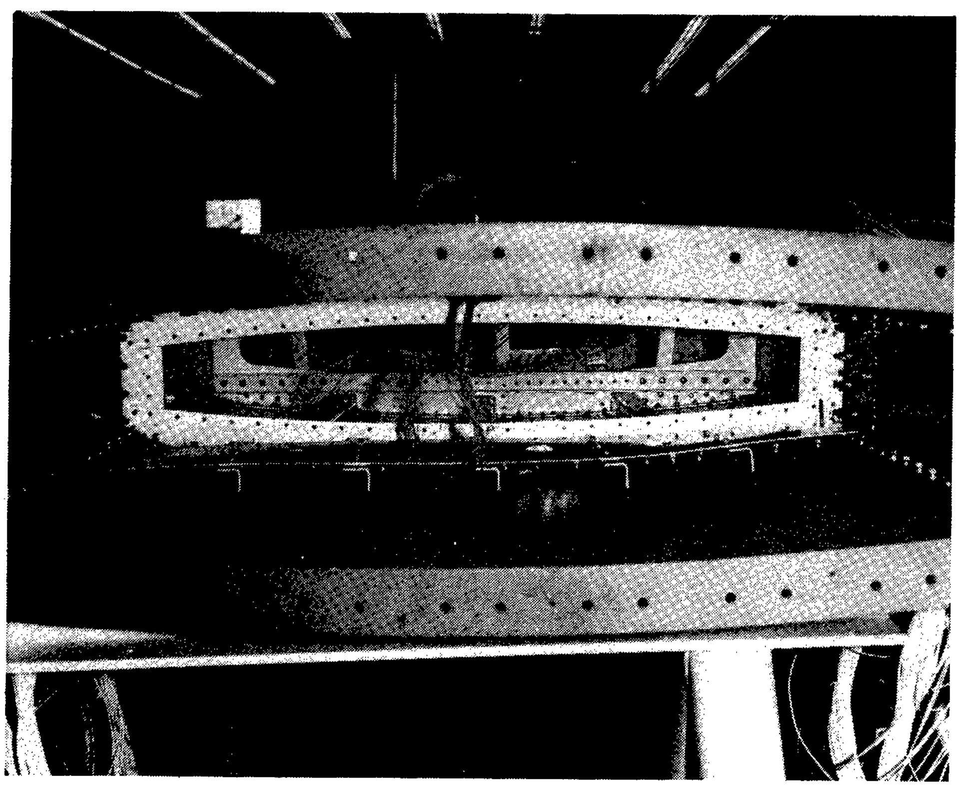


Figure 4.15. Interior View of Wing Box Showing Strain Gage Wiring.

4.5 PRESSURE TESTING

The sealing of the wing box was tested by the application of Argon gas pressure through one of the 6 ports in the steel end fittings. Pressure was monitored by means of a water manometer. Test pressure was 0.5 psig. Several leaks were found along the spars and these were repaired by the application of sealant to the inside of the specimen. Access was gained by removal of the access door. Testing and repairing continued until no leakage was evident. Pressure was raised briefly to approximately 1 psi with no evidence of leakage.

5

TESTING

5.1 INTRODUCTION

The wing box testing was accomplished at the Naval Air Development Center, Warminster, Pa. The planned static test sequence was as follows:

1. Bending to 120 percent limit
2. Torsion to 120 percent limit
3. Bending to 150 percent limit
4. Torsion to 150 percent limit
5. Combined bending and torsion to 120 percent limit
6. Combined bending and torsion to 150 percent limit
7. 150 percent limit pressure (10 psi)

The negative bending condition was not planned since it was not a critical load condition. For the combined condition, positive bending and torsion were applied without internal pressurization. Strain data were recorded for all load conditions. Deflection data were recorded for all conditions except internal pressurization. Data were recorded in 20% increments up to 100% L. L. and then in 10% increments thereafter.

5.2 INSTRUMENTATION

A total of 81 strain gages were installed on the wing box prior to shipment to the Navy. Six of these gages were inside the box and the remainder on the exterior. A total of 24 axial FAE-25-1253 gages were utilized and 19 FAER-25R-12-53 Rossette gages were installed. The gages were situated as indicated by Figures 5.1 through 5.5.

Table 5-1 gives the correlation between the strain gage number and the instrumentation channel number.

On the strain gage diagrams Figures 5.1 thru 5.5, all Rossette gages are indicated thusly ●, the "A" gage of each Rossette is parallel with the stringers, Gages B and C are read clockwise from Gage A. All interior gages were water proofed and cables

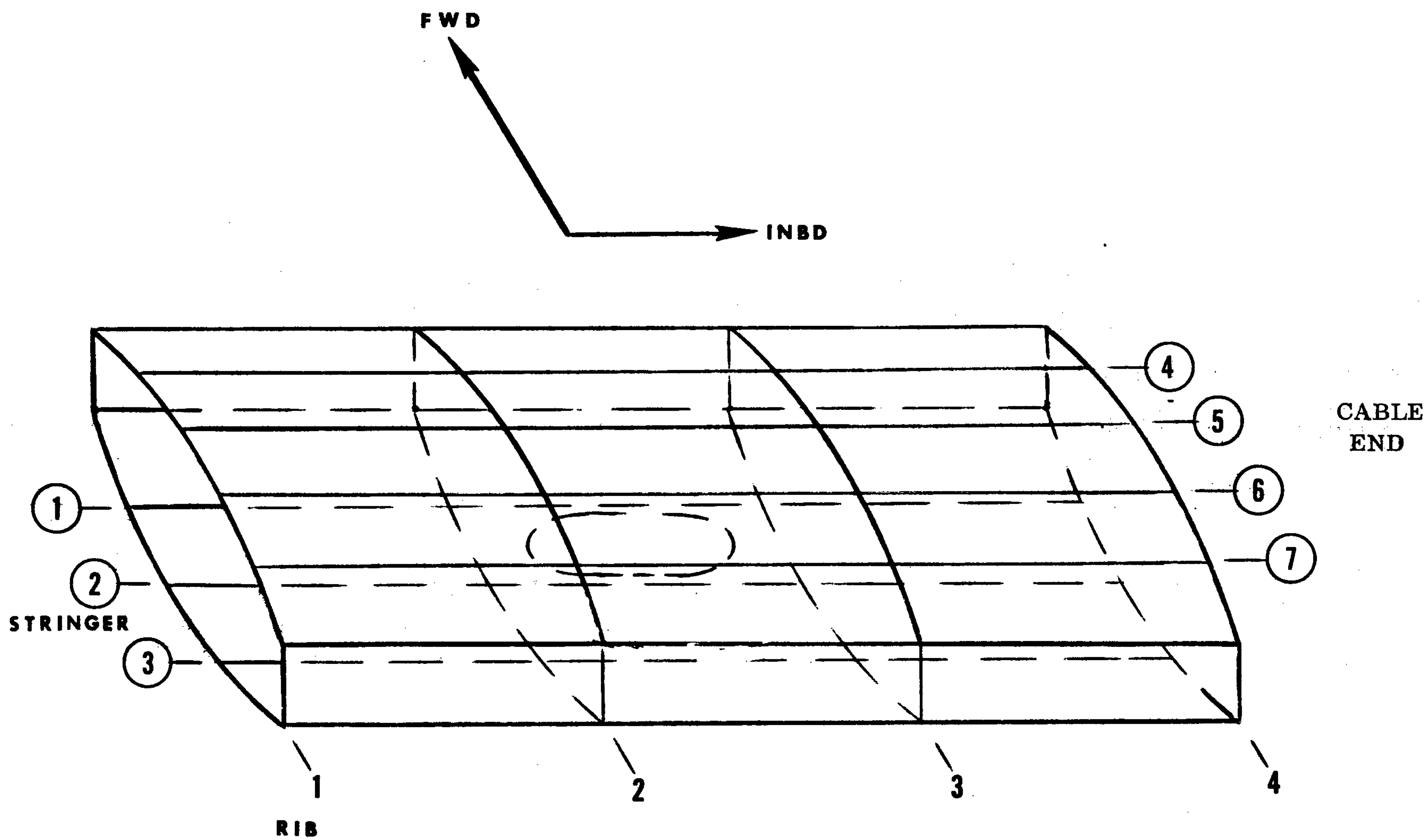


Figure 5.1. Orientation View
Wing Box Instrumentation.

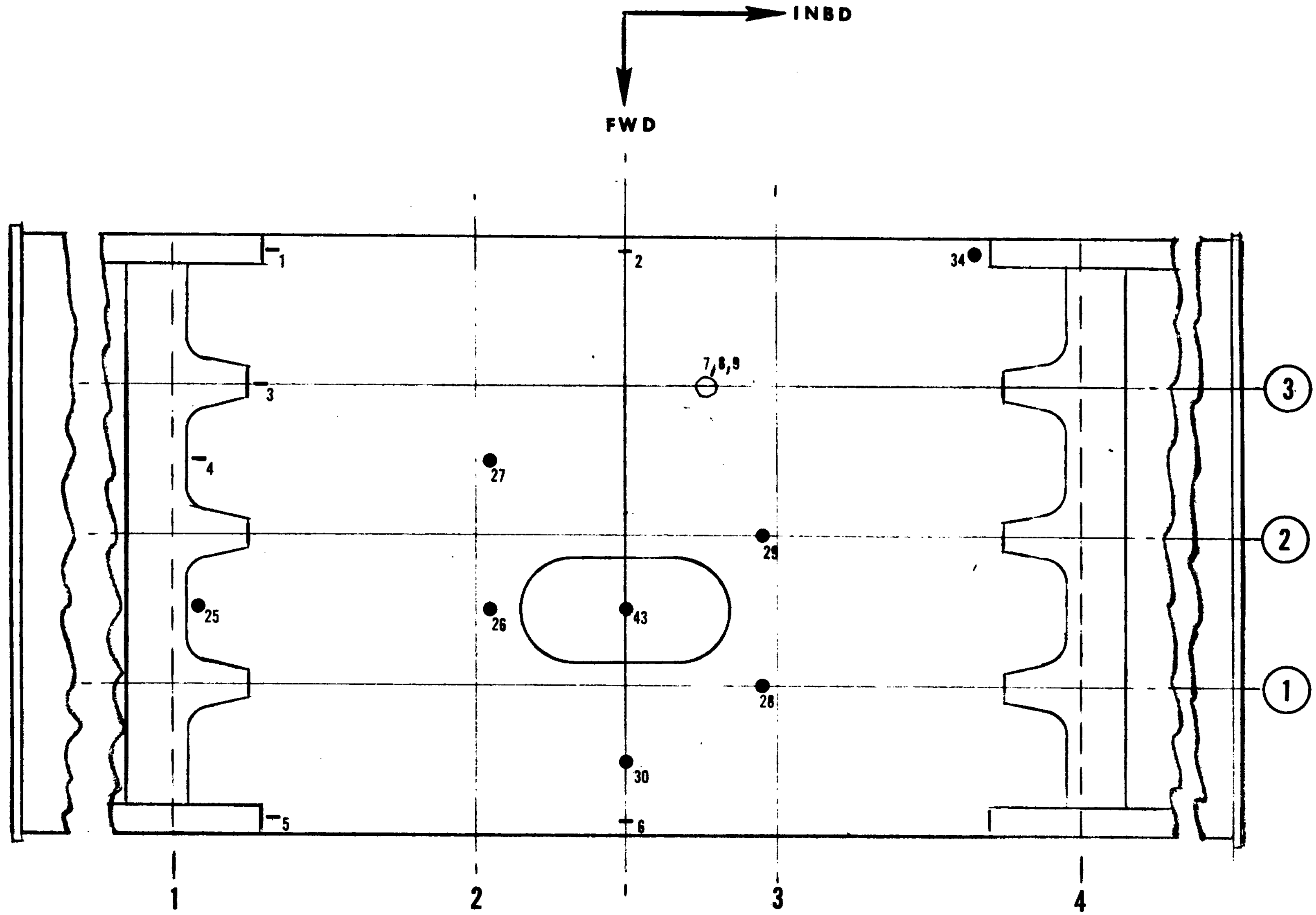


Figure 5.2. Wing Box Instrumentation.

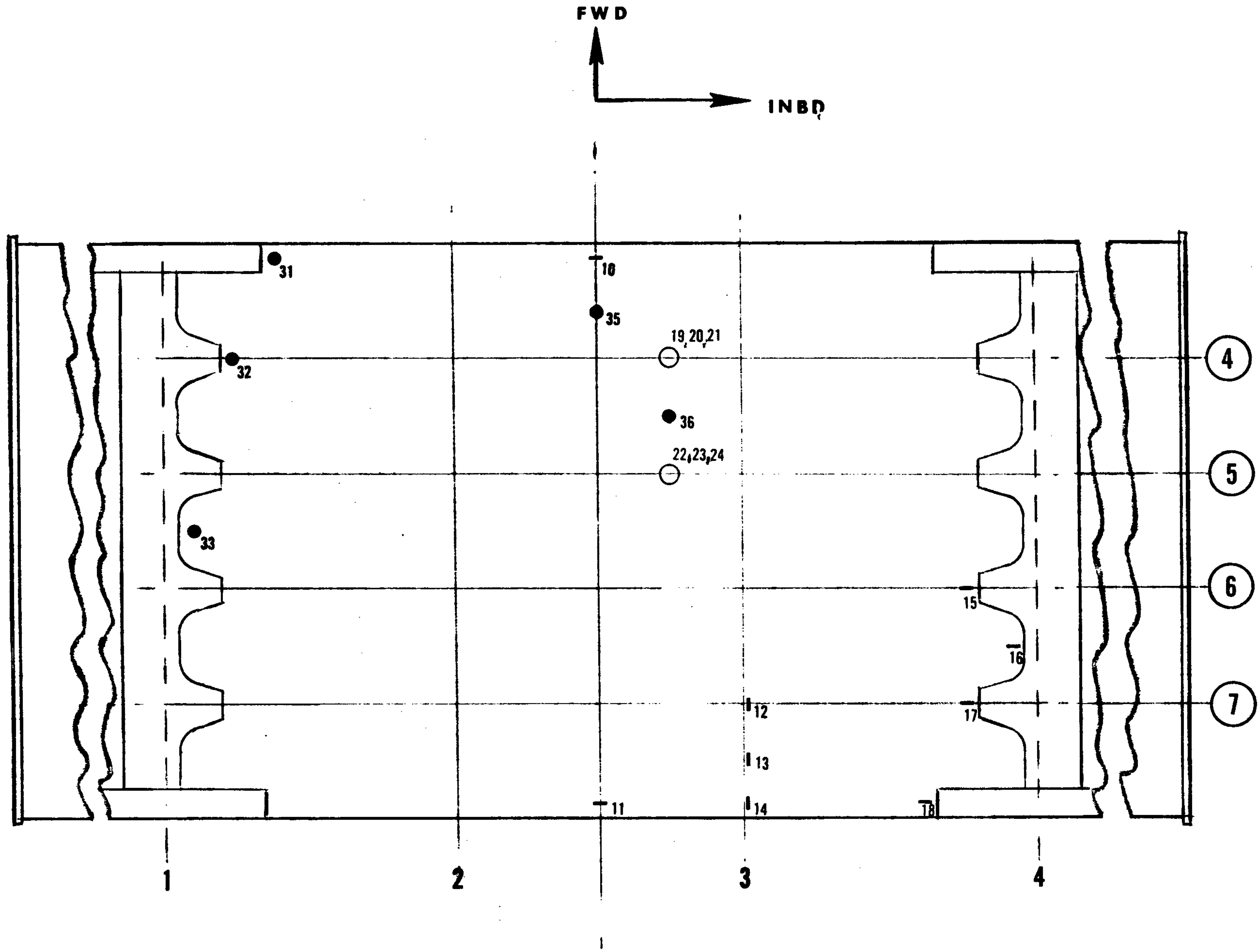


Figure 5.3. Wing Box Instrumentation.

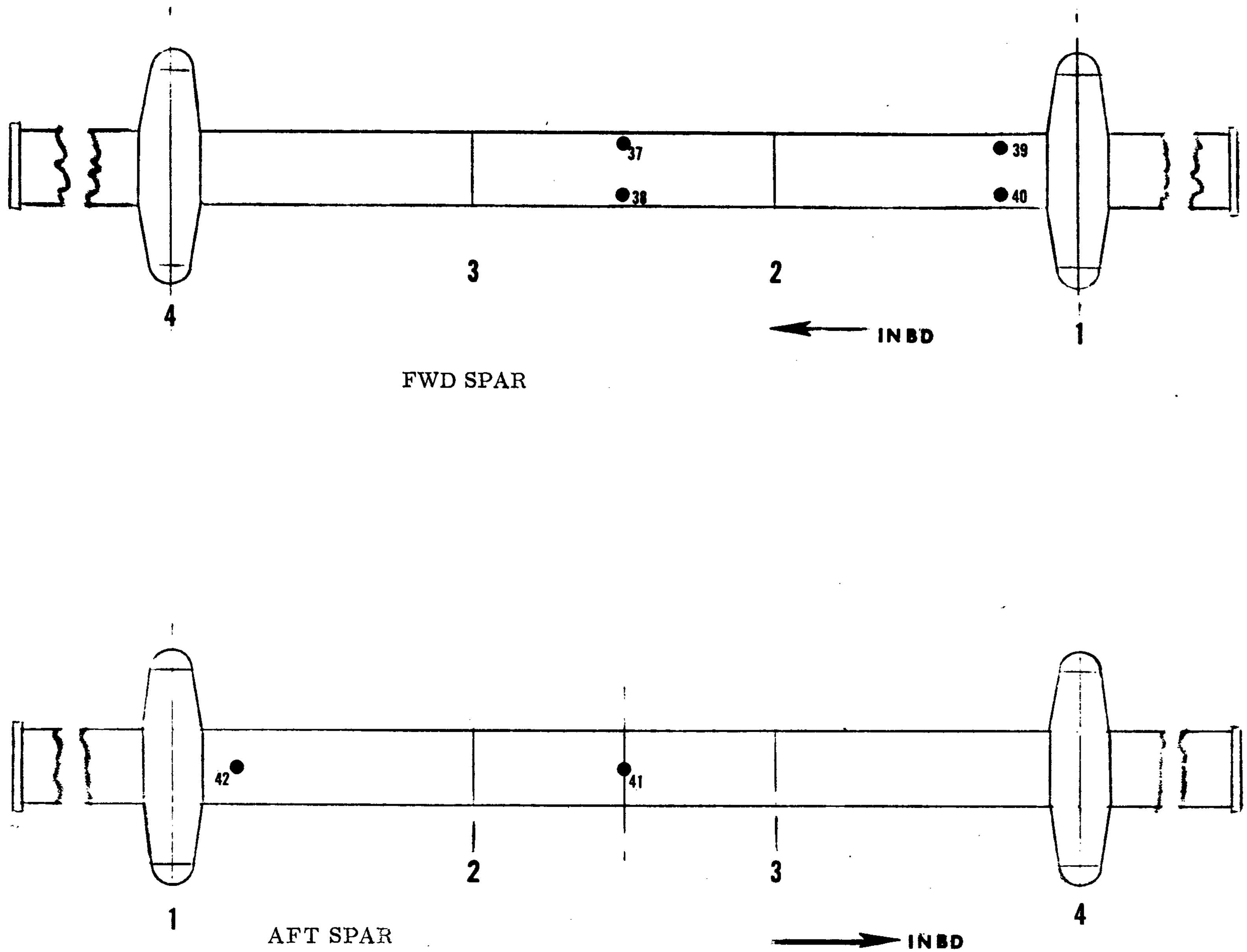


Figure 5.4. Wing Box Instrumentation.

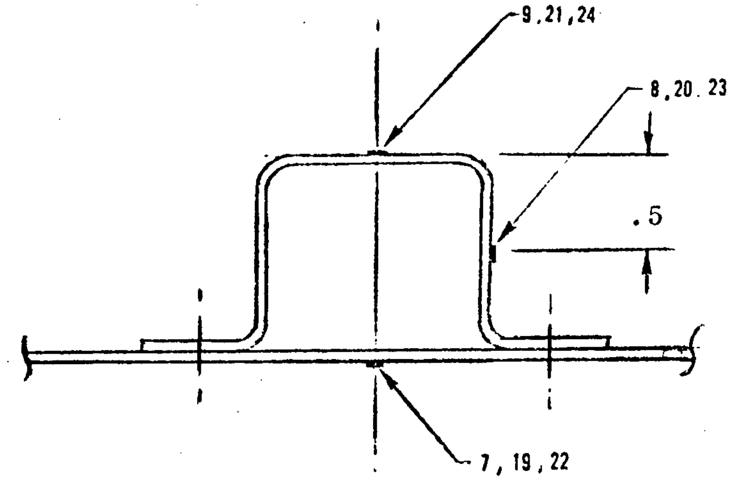


Figure 5.5. Stringer Cross Section Gage Location.

Table 5-1. Strain Gage-Channel Number Correlation

Gage	Channel	Gage	Channel	Gage	Channel
1	0	26a	27	35a	54
2	1	26b	28	35b	55
3	2	26c	29	35c	56
4	3	27a	30	36a	57
5	4	27b	31	36b	58
6	5	27c	32	36c	59
7	6	28a	33	37a	60
8	7	28b	34	37b	61
9	8	28c	35	37c	62
10	9	29a	36	38a	63
11	10	29b	37	38b	64
12	11	29c	38	38c	65
13	12	30a	39	39a	66
14	13	30b	40	39b	67
15	14	30c	41	39c	68
16	15	31a	42	40a	69
17	16	31b	43	40b	70
18	17	31c	44	40c	71
19	18	32a	45	41a	72
20	19	32b	46	41b	73
21	20	32c	47	41c	74
22	21	33a	48	42a	75
23	22	33b	49	42b	76
24	23	33c	50	42c	77
25a	24	34a	51	43a	78
25b	25	34b	52	43b	79
25c	26	34c	53	43c	80

were brought out through the large port on the inboard end. Gages on stringers were located in accordance with Figure 5.5 and are indicated thusly ● on the overall views.

A total of 14 deflection measurements were taken during most of the testing. The deflection gages were located as indicated in Figure 5.6.

5.3 TESTING

The test setup is shown in Figure 5.7. The compression surface is the lower surface. The specimen was simply supported at both ends in bending; the inboard end had two supports to resist the torque. The outboard support was free to twist about its center-line and also to move inboard under specimen deflection. For bending, the loads at the two ends of the test section were applied by one load actuator which was distributed to four load points. For torsion, two loads, which form a couple, were applied at the outboard end plate. For the combined condition, both bending and torsion loads were applied simultaneously.

5.3.1 PURE BENDING TESTS. The wing box was subjected to bending loads to 120 percent of limit and to 150 percent of limit without failure. The compression skin buckled visibly above 80 percent limit load.

The unloaded wing box is shown in Figure 5.8. The compression skin is shown to be smooth and free of distortion. Figure 5.9 shows the wing at 120 percent limit load in bending. The skin buckles are apparent in all three bays. Figure 5.10 is a plot of 3 compression strain gages during the first bending test to 120 percent limit. The loading was interrupted after the initial 110 percent loading. The second group of curves between 100 and 120 percent limit represent the strain when the test was resumed.

Gages 36A and 35A were on the skin between stringers while Gage 20 was on the vertical hat section of the first stringer. Although buckles were visible in the skin at 120 percent limit the strain gages did not show any great deviation from linearity.

Figure 5.11 shows four similarly placed gages during the 150 percent limit cycle. There were no back to back skin gages, however the buckling of the skin is apparent as is the increasing load acquired by the stringer.

Figure 5.12 was plotted to determine the uniformity of strain at the mid point of the beam during the ultimate loading test. There was not equal strain in each cap, but the average compression strains were found to be almost exactly equal to the tension strain. Gages 10 and 6 represent a couple at one spar while gages 11 and 2 a couple at the other. No logical explanation for the differences has been found.

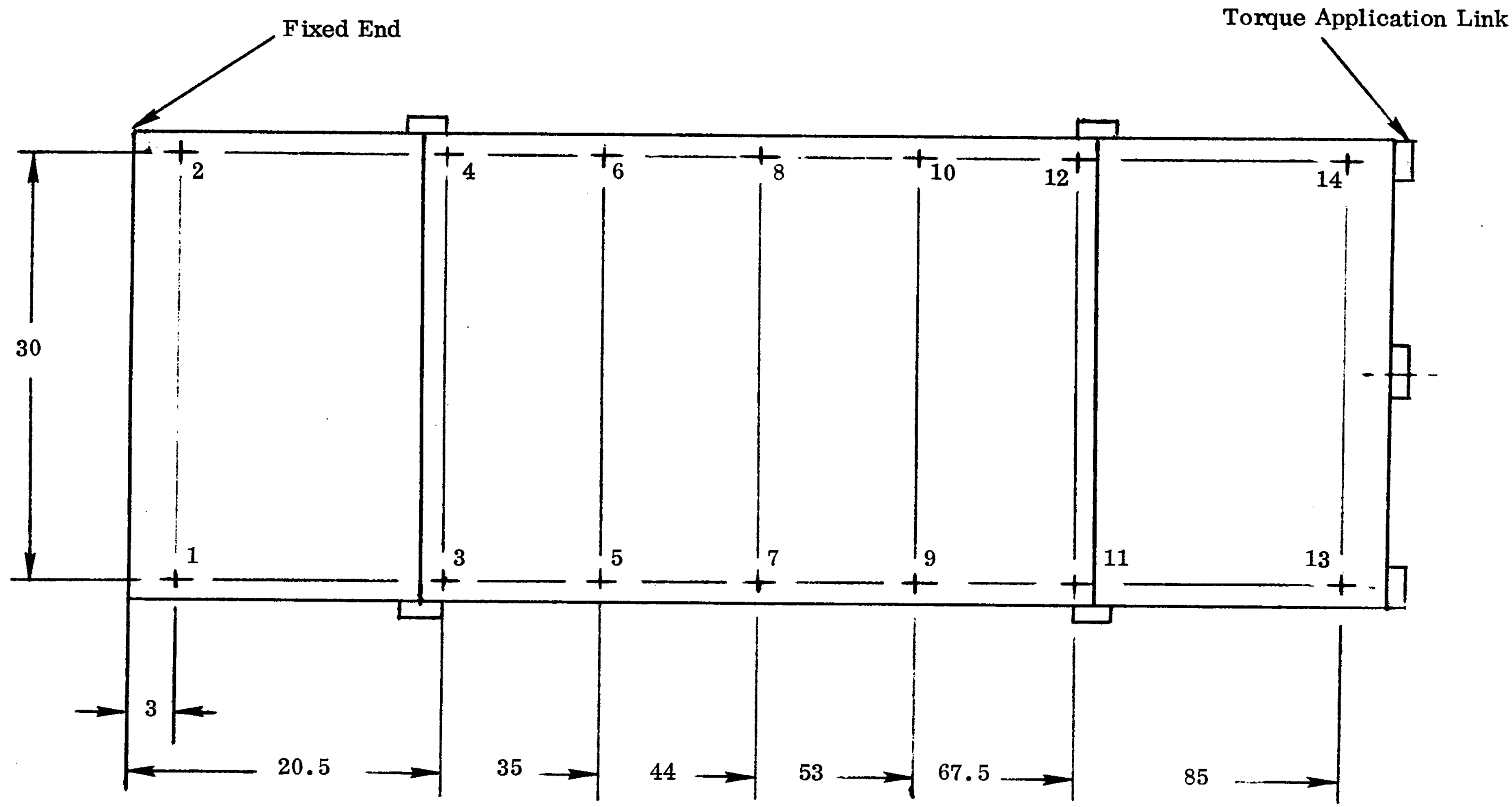


Figure 5.6. Location of Deflection Measurements Viewed Looking Down

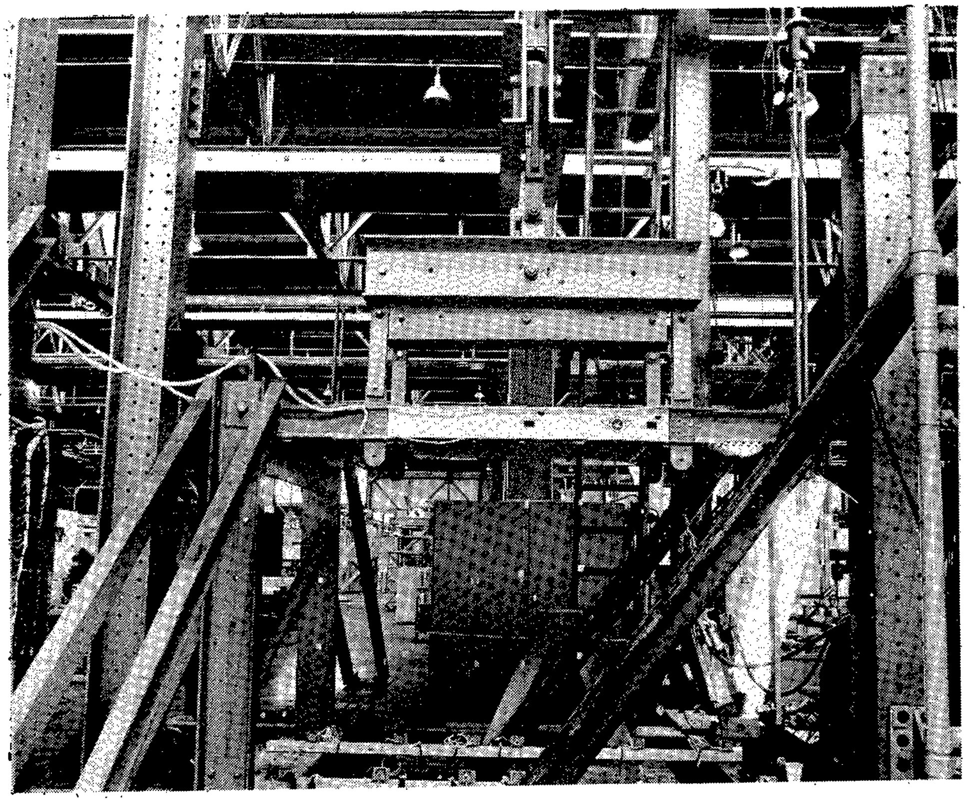


Figure 5.7. Overall View of Wing Box in Test Fixture

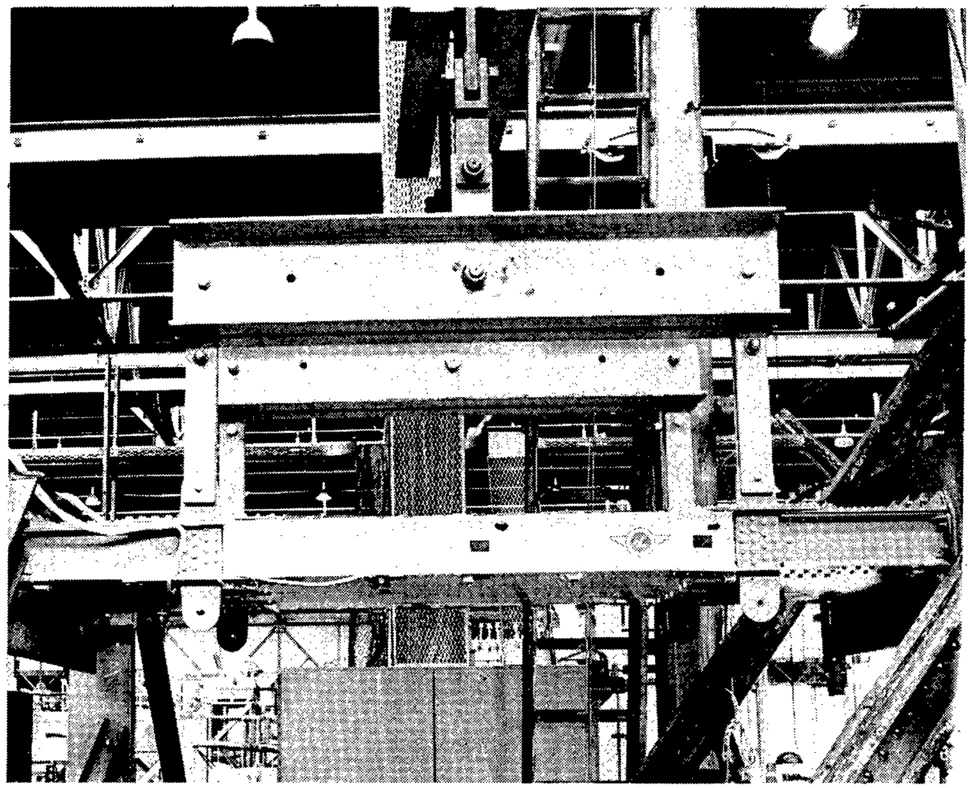


Figure 5.8. Overall View of Wing Box in Test Fixture with No Load.

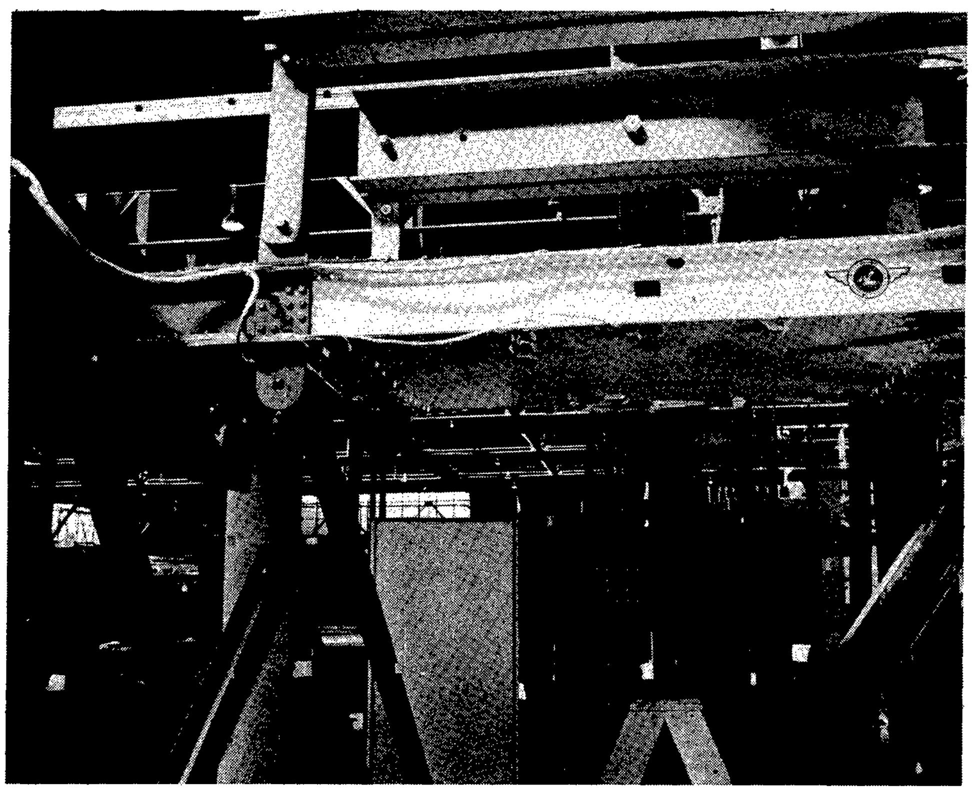


Figure 5.9. Buckles in Wing Compression Skin Due to Pure Bending.

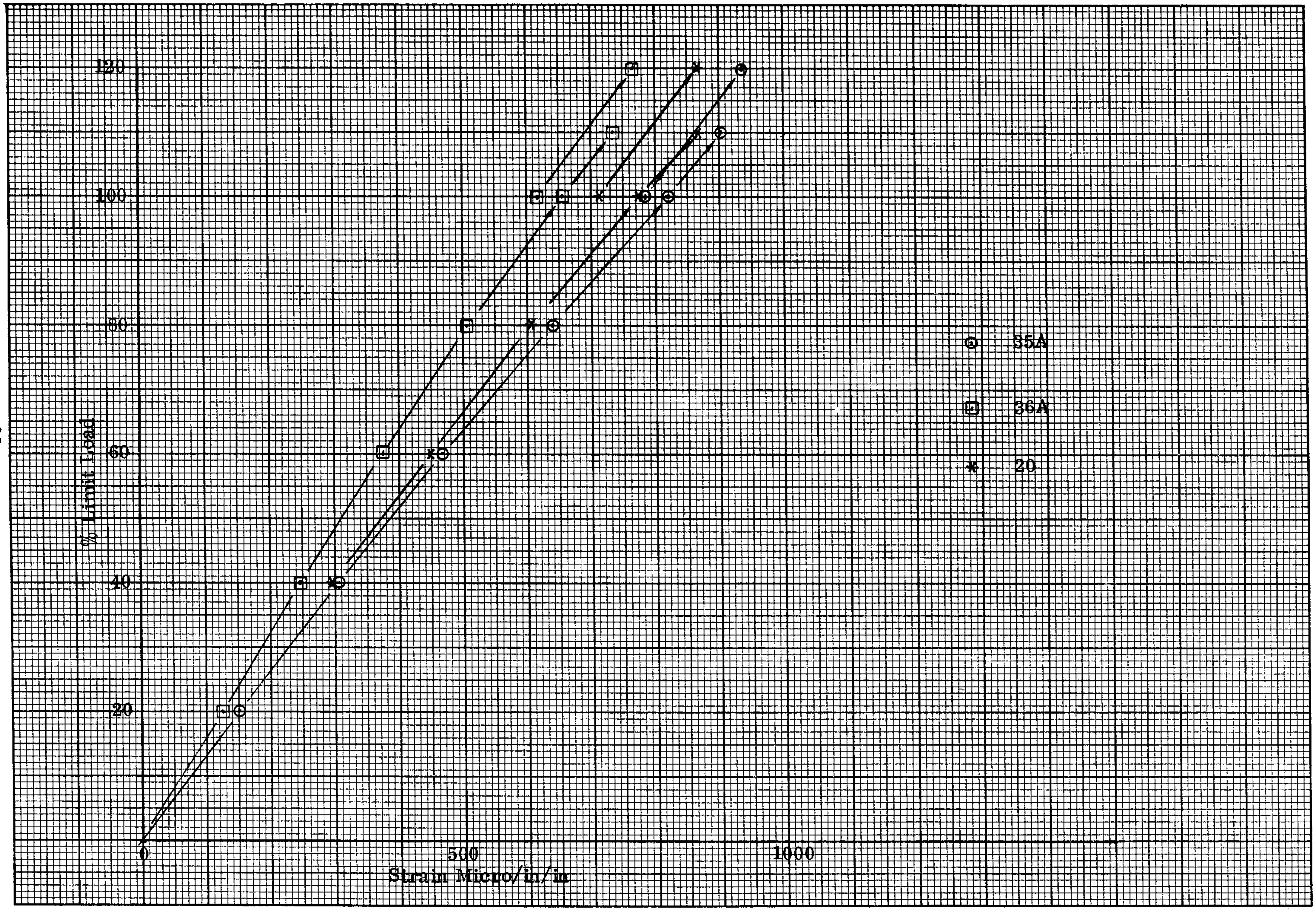


Figure 5.10. Compression Strains-Wing Box Bending.

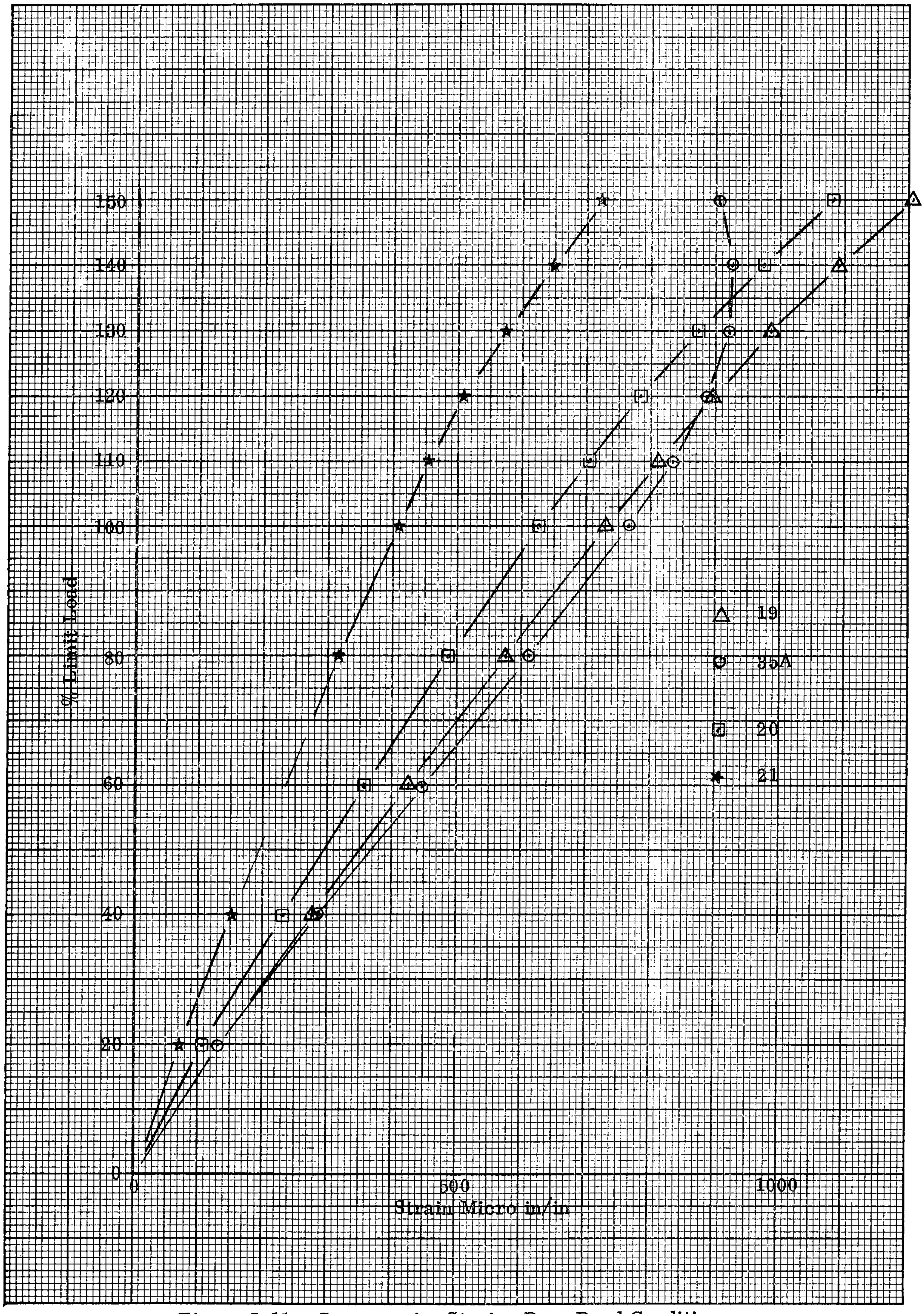


Figure 5.11. Compression Strains-Pure Bend Condition.

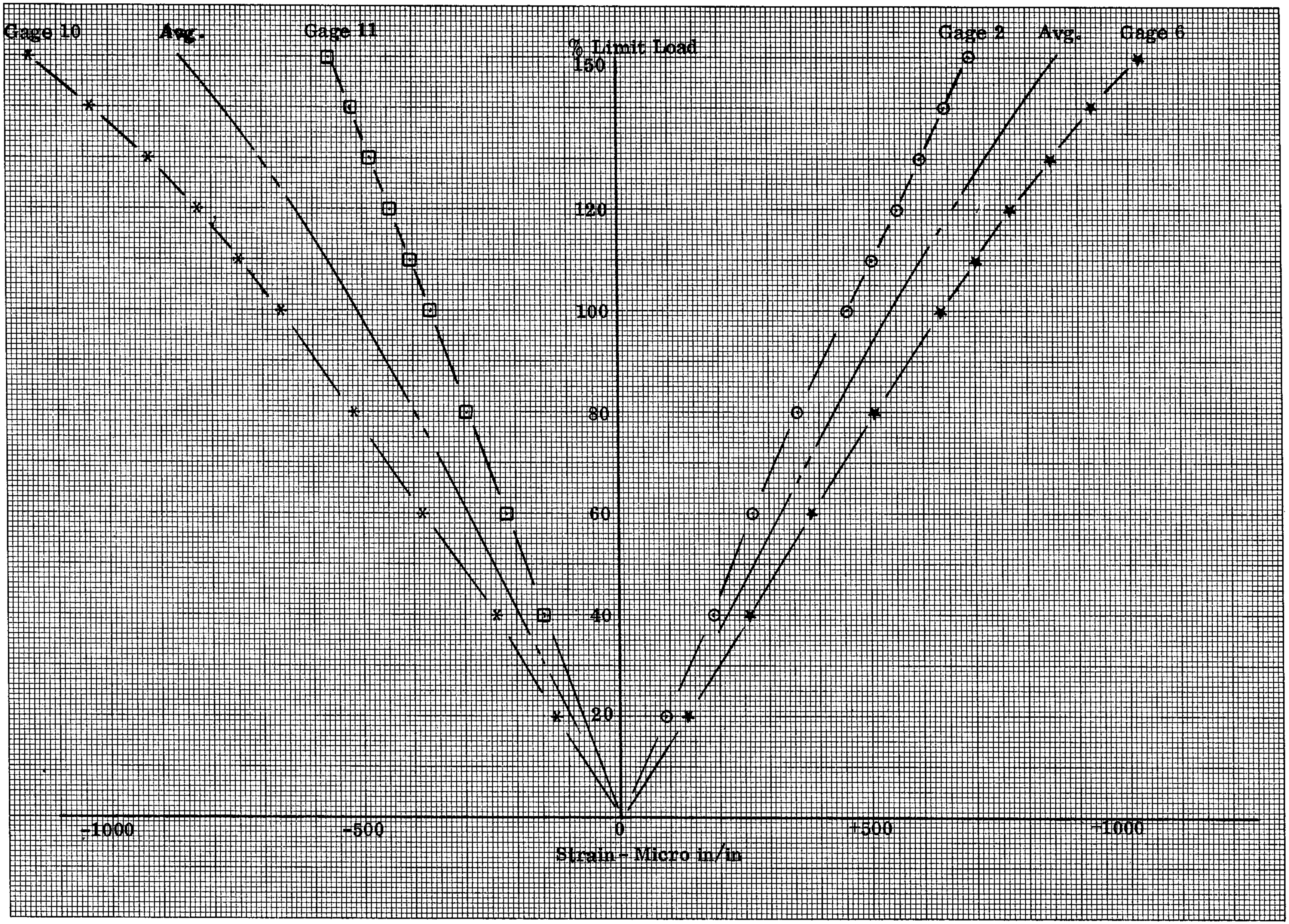


Figure 5.12. Spar Cap Strain-Pure Bending.

5.3.2 <u>TORSION TESTING</u>. The wing box was tested in torsion in two series of tests. The first to 120 percent of limit and the second to 150 percent of limit load. Figures 5.13 through 5.15 show the progression of diagonal tension wrinkles on the lower skin of the wing box as it was loaded to 120 percent LL. The buckles did not grow in amplitude appreciably as the load intensity increased. Figures 5.16 and 5.17 show the strain reading on the top and bottom skin during the initial loading cycle. Figure 5.18 shows a comparison of the strain readings between the 110 percent LL series and the 150 percent LL series of tests. The wing box was usually inspected after the testing and no evidence of failure or damage was evident.

5.3.3 <u>COMBINED LOADING TESTS</u>. The wing box tests in combined loading of torsion and bending were also performed in series. The first up to 120 percent LL and the second to 150 percent LL.

As indicated in Section 2.2.1 the torsion load which combines with the bending moment is much smaller than the pure torsion mode. The two series of tests were performed and no failure or distress of the wing box was apparent. Figures 5.19 to 5.22 are a series of photographs showing the compression skin buckles from 80 percent LL to 150 percent LL. No damage was evident after this testing.

5.3.4 <u>PRESSURE TESTING</u>. The wing box was next to be tested in 10 psig pressure equal to 150 percent limit loading. The wing was filled with water and pressurized by air. No damage was apparent from the pressure testing but it did reveal that leaks had developed as a result of the structural testing. The predominant leaks were along the spar caps between the skin and spar cap but were not of sufficient size to prevent pressurization of the box.

5.3.5 <u>FATIGUE TESTING</u>. The wing box was next subjected to spectrum fatigue cycling using Spectrum A of Military Specification MIL-A-8866 for the combined condition. The specimen was assumed to be suitable for a 7.5g fighter with a lifetime of 6,000 flight hours. Loads were applied for positive limit load levels from 1.0g to the appropriate limit load level. Only load levels of 75% LL and above were applied since previous evidence with this material indicated no appreciable damage caused by load levels below the buckling load. Loads were applied in 20-hour blocks. Cycling was continued through two lifetimes (12,000 hours) without any signs of damage or permanent set.

Having sustained the required static load conditions and specified lifetime, the decision was made to determine the number of lifetimes the specimen was capable of sustaining. Consequently, cycling was continued; however, the torsion loads were increased to two times those previously applied to obtain a more realistic combined

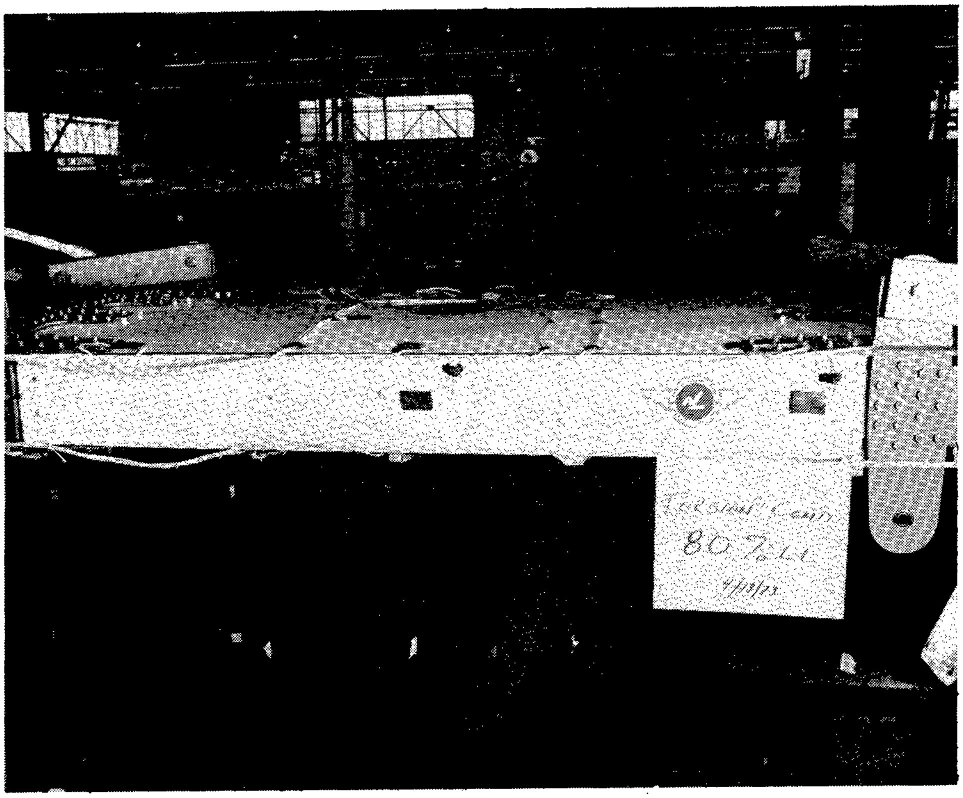


Figure 5.13. Wing Box at 80% L.L. in Torsion.

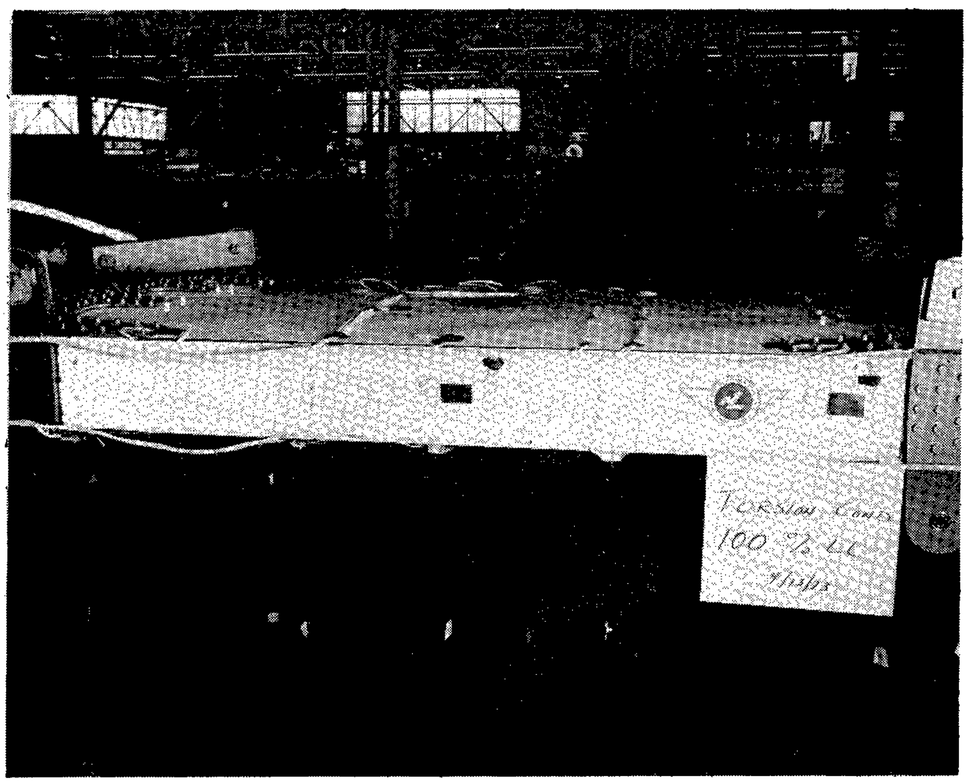


Figure 5.14. Wing Box at 100% L.L. in Torsion.

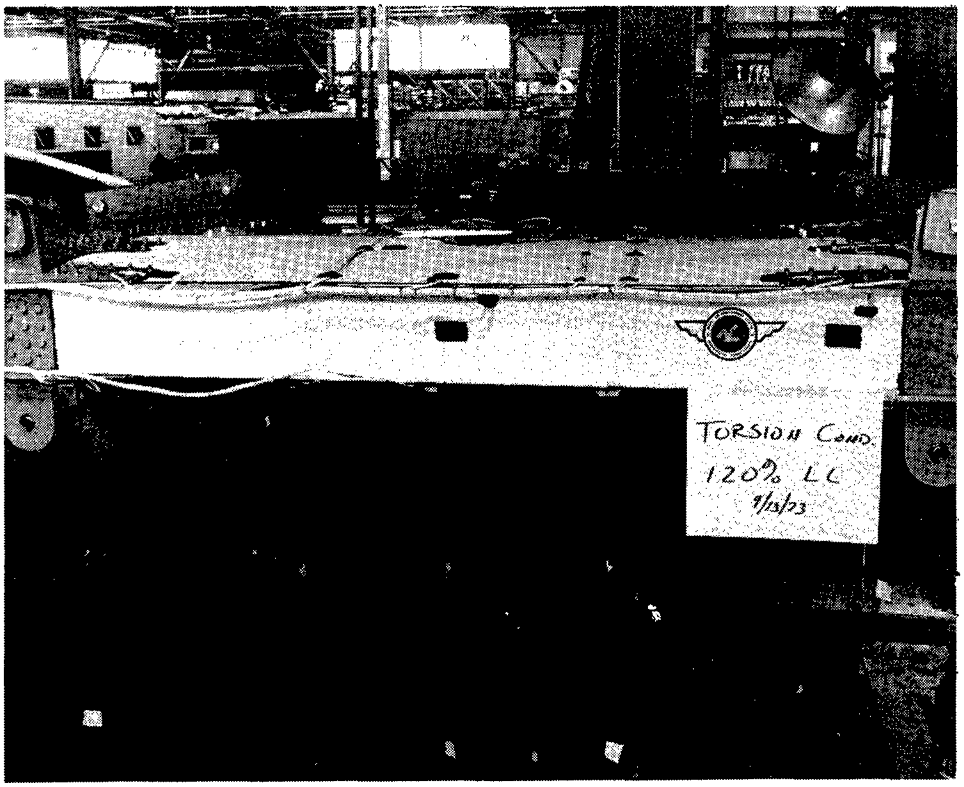


Figure 5.15. Wing Box at 120% L.L. in Pure Torsion

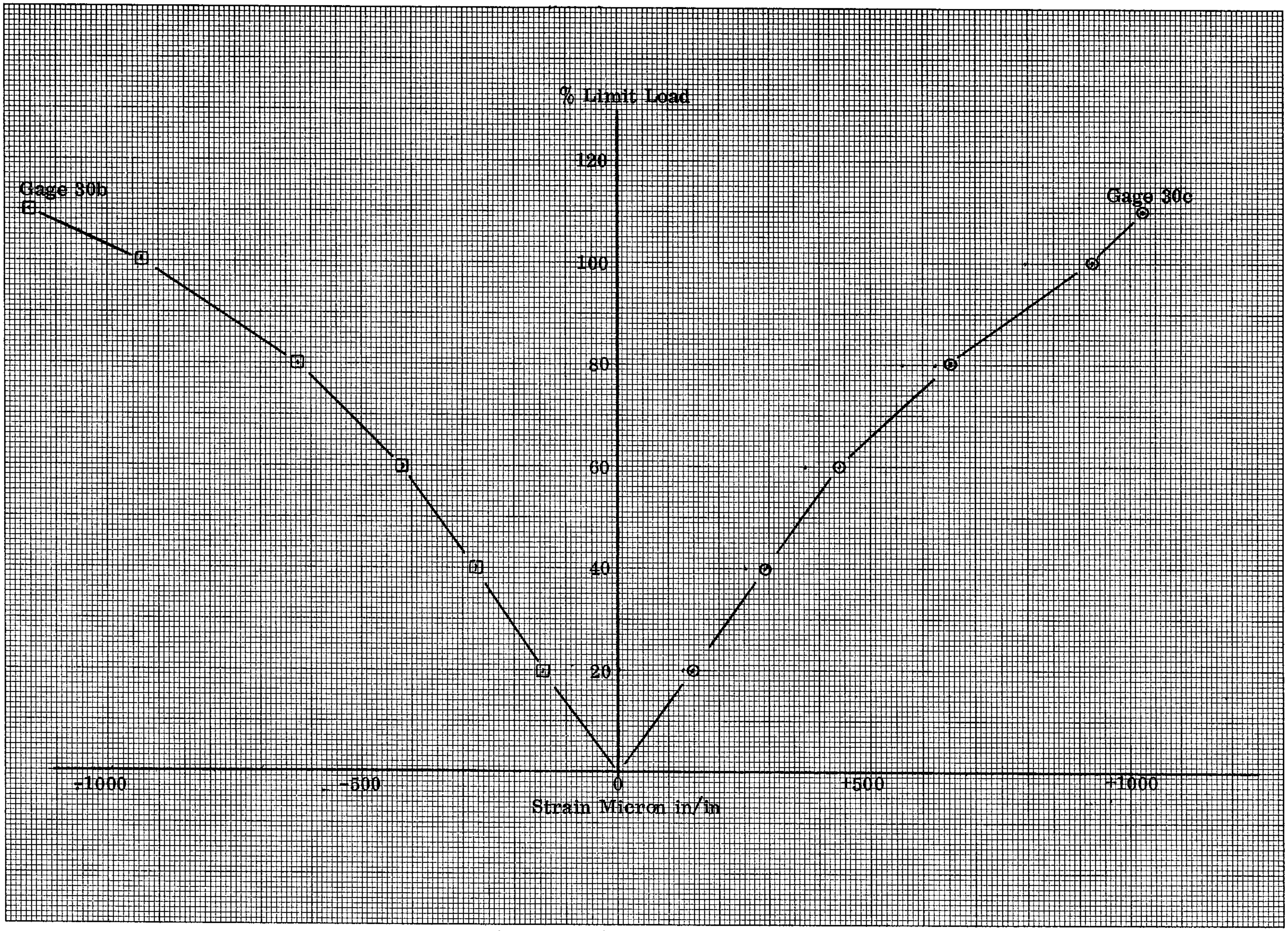


Figure 5.16 Upper Surface Strains-Torsion Condition.

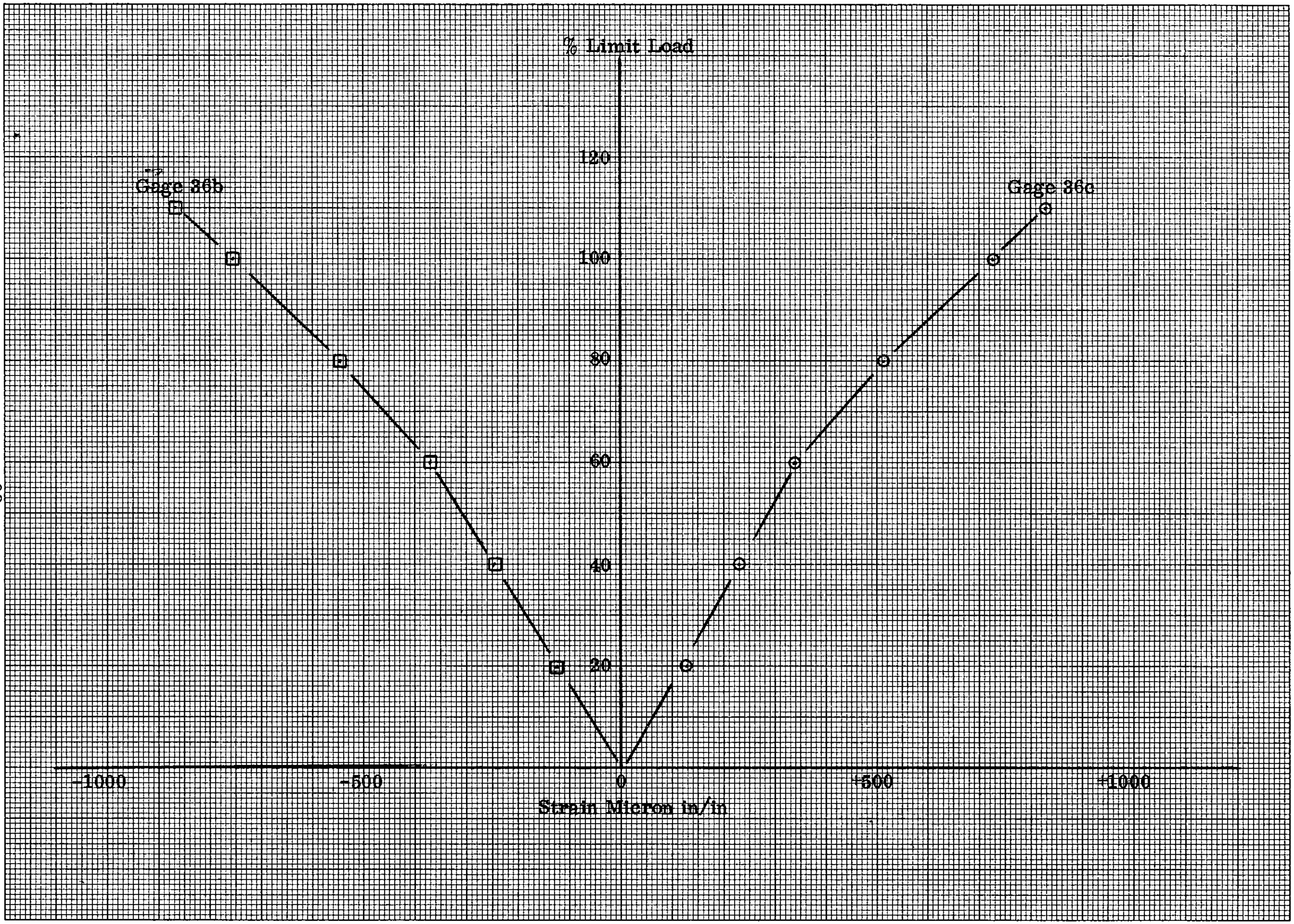


Figure 5.17. Lower Surface Strain-Torsion Condition.

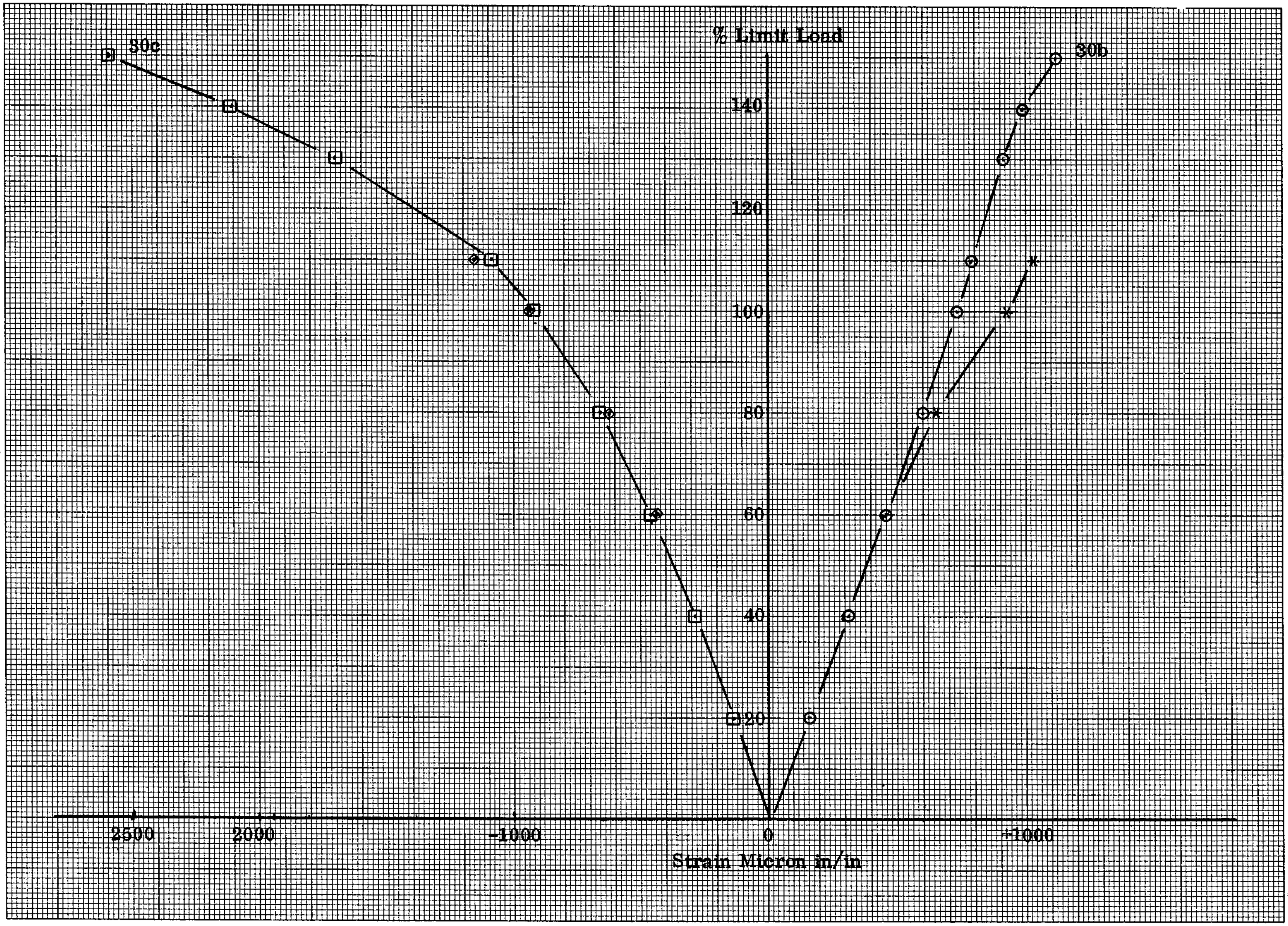


Figure 5.18. Comparison of 110% and 150% L. L. Skin Strains-Torsion Case.

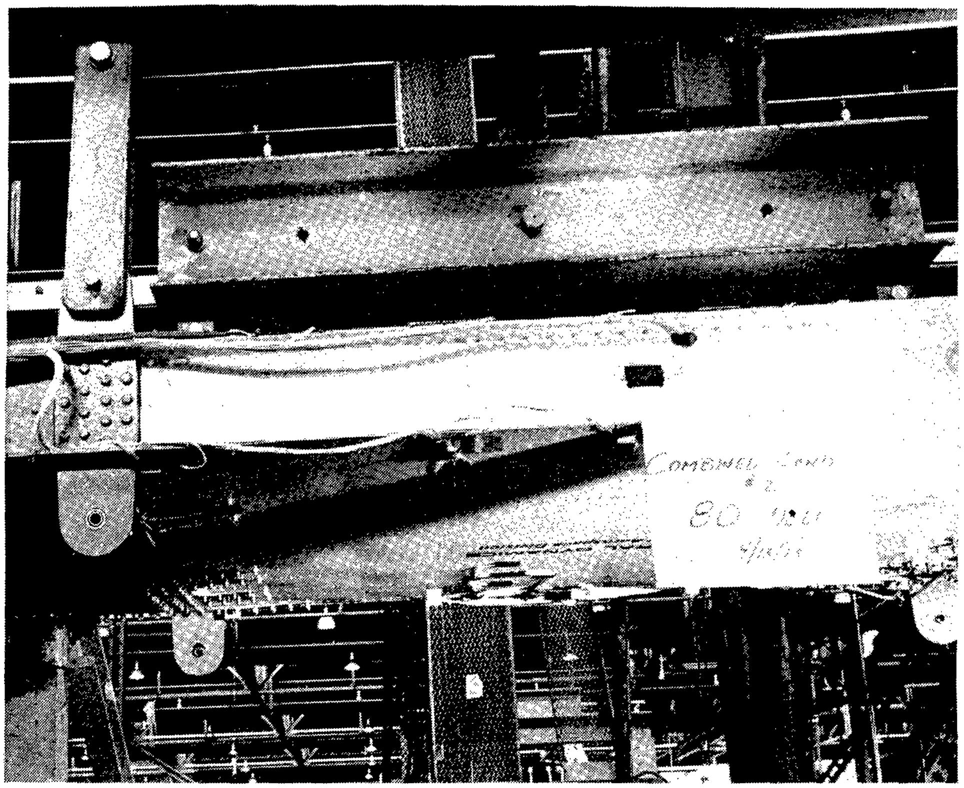


Figure 5.19. Compression Skin Buckles at 80% L. L.

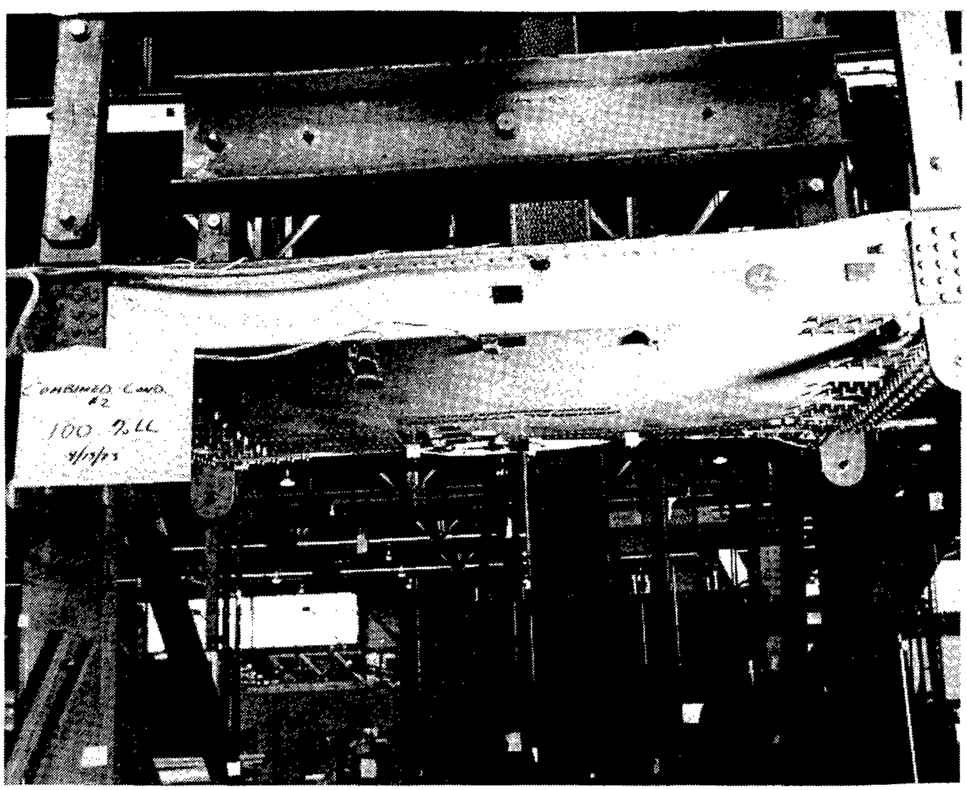


Figure 5.20. Compression Skin Buckles at 100% L. L.

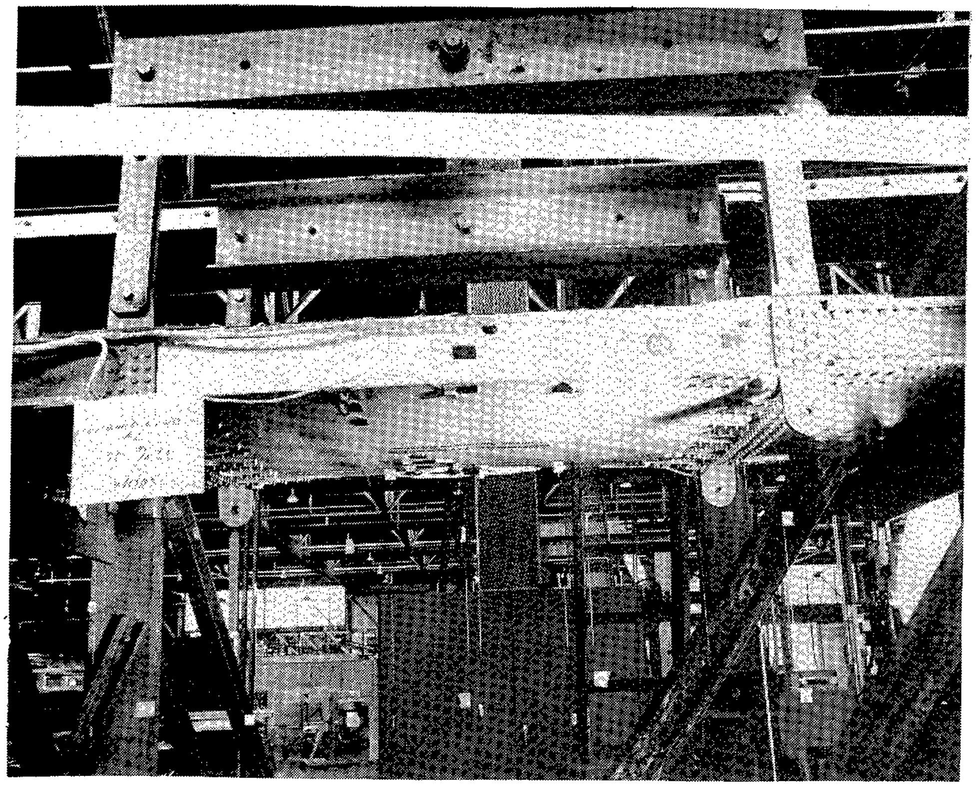


Figure 5.21. Wing at 130 Percent Limit Load Combined Loading Case

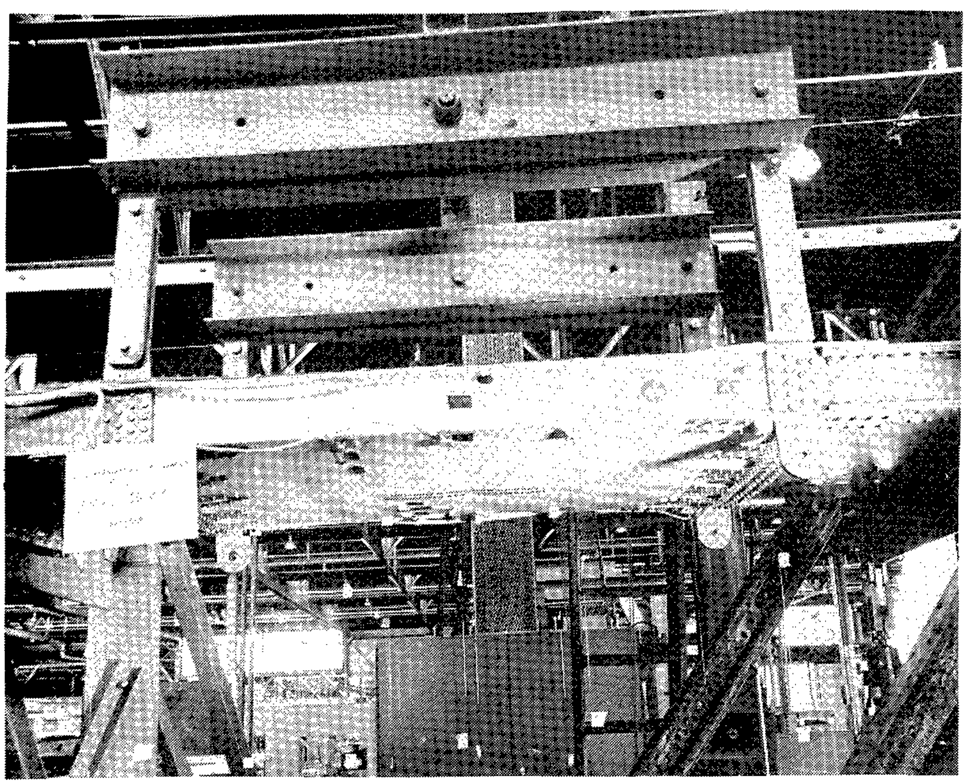


Figure 5.22. Wing at 150 Percent L. L. Combined Load Case

loading condition. Failure was observed during Block 1580 (31,600 hours) which is over five lifetimes. Failure occurred as buckling of the compression surface and subsequent inter-rivet cracking along two stringers as shown in Figures 5.23 and 5.24.

Removal of the inspection door permitted the interior of the wing to be inspected and revealed that the cause of the buckle was the fatigue failure of several of the stringers along the radius between the skin flange on the side of the hat (Figure 5.25). This condition allowed the skin and skin flange to buckle independently from the remainder of the stiffener. The skin had also cracked along the rivet rows in some places. All these fatigue cracks were due to the flexure in the skin flanges of the stringers. These areas were not instrumented so the magnitude of the buckling strain is not available.

5.3.6 <u>REVERSE BENDING</u>. Since no damage was apparent on the tension surface, a static test was performed to compare the buckling load of the tension surface (upper skin) with the predicted buckling load. The PUL (Predicted Ultimate Load) for buckling of the upper skin was 522,000 in/lb. This is considerably higher than the design ultimate load in Paragraph 3; however, the excessive margin is attributed to a relatively cursory design effort in which three stringers were considered more than adequate to sustain the negative bending load. The wing box was inverted in the test jig so that what was previously the tension surface was now the compression surface. The loads were applied in 20% increments of design limit load. Buckling of the skin was noted at approximately 42% PUL. Loading was continued to 86% PUL at which point the load was removed to determine if any permanent set was evident in the form of skin buckles. None were apparent so the wing box was loaded slowly again to failure. Failure occurred at 93% PUL as buckling of the panel across the width of the wing box as shown in Figures 5.26 and 5.27. Examination of the stringers indicated that the vertical portions of the hat section had failed in a local mode of buckling.

When the compression skin buckled, the wing still carried the applied load even through portions of the inboard flange of the spars had buckled with the skin. It has been estimated that the remaining spar caps are carrying over 300,000 psi in this condition.

The predicted failing load agrees favorably with the static test results considering that the buckling strength may have been reduced due to the fatigue cycling. This agreement shows that the methods used for the design of the compression skin are valid and are consistent with good design practice. Since the compression panel was designed with reasonable margins of safety and the fatigue failure occurred in the compression panel, the fatigue life attained is considered representative of what might be expected for structures of similar material and construction.

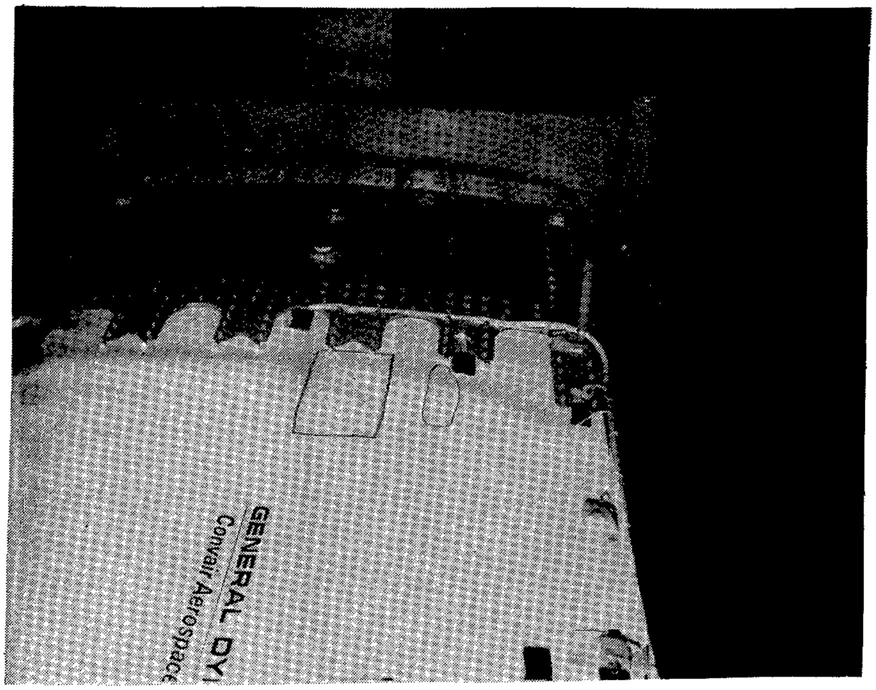


Figure 5.23. Buckle in Compression Skin After Fatigue Testing.

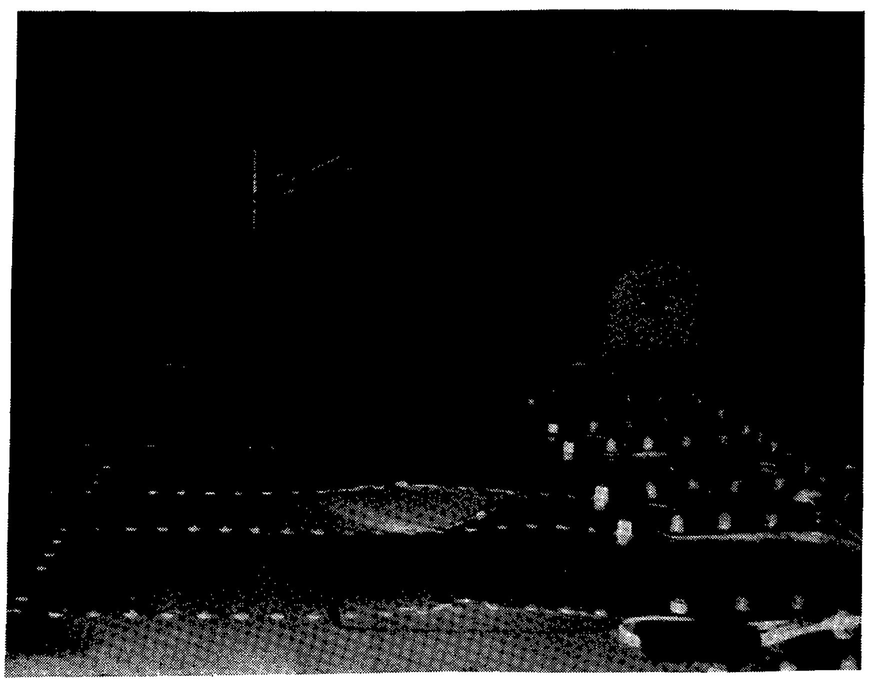


Figure 5.24. Crack in Skin at Stringer Attachment Rivets After Fatigue Testing.

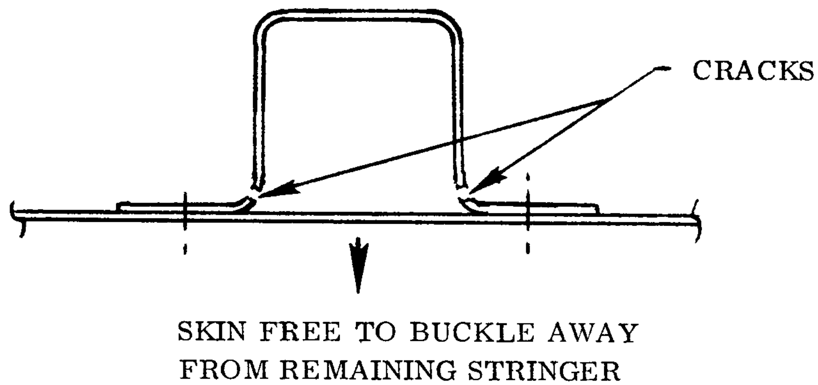


Figure 5.25. Detail of Cracked Stringer.

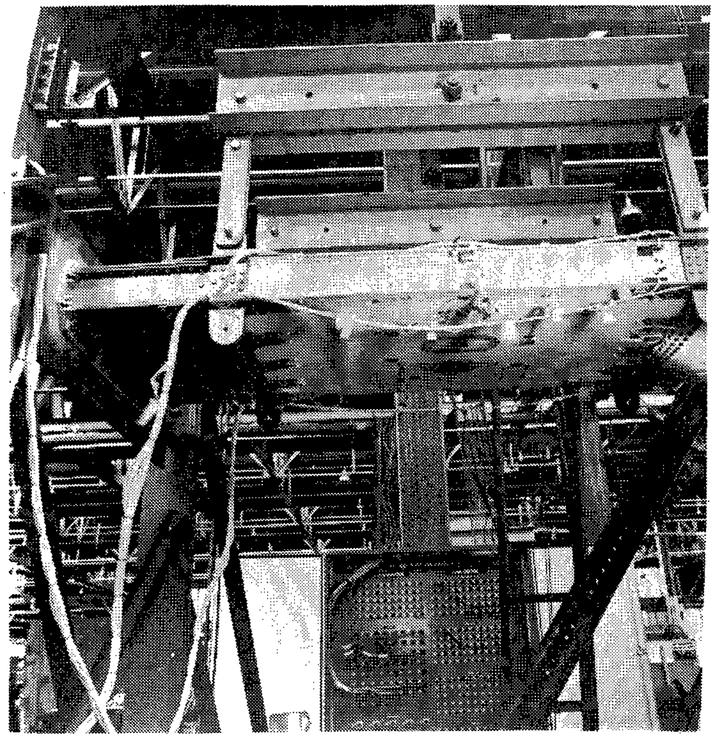


Figure 5.26. Buckle in Compression Skin After Failure in Reverse Bending.

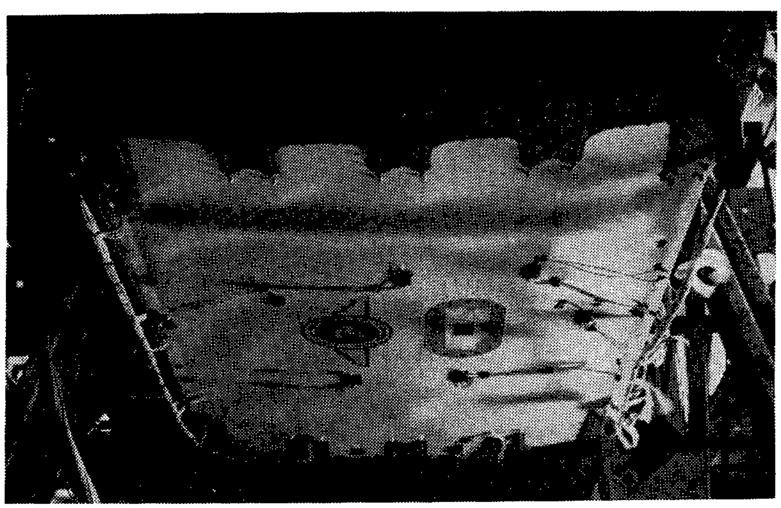


Figure 5.27. Buckle in Compression Skin Reverse Bending.

6

CONCLUSIONS AND RECOMMENDATIONS

6.1 INTRODUCTION

A wing box specimen was designed, analyzed and fabricated using boron/aluminum as the primary structural material. A weight saving of 24.8% was realized over an all aluminum wing designed to the same criteria.

The wing box was structurally tested and met or exceeded all static load requirements.

Although not a design requirement, the wing was subjected to the fatigue spectrum for Navy fighter aircraft and successfully passed two 6,000 hour lifetimes without failure. Additional fatigue testing eventually caused failure of the compression skin locally after 31,000 equivalent flight hours.

The wing was finally tested to destruction in reverse bending with failure occurring at 94% of the predicted failing load which was in excess of the design ultimate load.

6.2 DESGIN AND ANALYSIS

6.2.1 The testing reflected the conservative approach used in the design and analysis of the wing. Using more accurate analysis methods and more realistic material allowables (which were not available when the wing was designed) increased weight savings could be achieved.

6.2.2 Load introduction from the steel end fittings was accomplished using mechanical fasteners. Great care was exercised in the design and analysis of these joints which demonstrated that careful attention to design details and the use of a fine grid finite element model will result in reliable mechanical joints in composite structures.

6.2.3 At the time the wing box was designed a good failure and design criteria for boron/aluminum did not exist which resulted in the use of conservative assumptions being used. This is particularly true of panels subjected to biaxial strains and fatigue loading. A more detailed knowledge of the behavior of the material is needed in order to permit less conservative designs to be made.

6.2.4 There is a need to develop improved buckling and crippling information for design allowables prediction for crossplied boron/aluminum. Both flat plate and curved panel data is needed.

6.3 FABRICATION

This program demonstrated that boron/aluminum structures can be fabricated with todays technology and existing shop equipment and personnel. Using sheet metal fabrication techniques, these structures can be fabricated at reasonable cost.

REFERENCES

1. Kiger, R. W., "Graphite Epoxy Wing Box Development," GDC-ERR-1469, Convair Aerospace, August 1973.

2. Spier, E. E., "Crippling Analysis of Unidirectional Boron/Aluminum Composites in Compression Structures," Convair Aerospace Division of General Dynamics, September 1972.

APPENDIX A

SPECIFICATION FOR BORON-ALUMINUM SHEET MATERIAL

SHEET, COMPOSITE, BORON FILAMENT, ALUMINUM ALLOY SPECIFICATION FOR BORON ALUMINUM WING BOX

1. SCOPE

1.1 Scope. This specification establishes the requirements for a composite boron filament aluminum alloy material.

1.2 Classification. The material covered by this specification shall be classified in the following types and grades.

Type I unidirectional plies (filament)

Type II cross plies (filament)

Grade A - 47.5 ±2.5 per cent boron filament by volume

Grade B - 45.0 ±2.50 per cent boron filament by volume

1.3 Classification identification. For classification of the material covered by this specification see 6.3.

2. APPLICABLE DOCUMENTS

2.1 Unless otherwise specified below, the following documents of the issue in effect on date of Convair's request for quotation form a part of this specification to the extent specified.

SPECIFICATIONS

Federal

QQ-A-250	Aluminum and Aluminum Alloy Plate and Sheet; General Specification For
FTMS No. 151, Method 211.1	Metals; Test Methods
Fed. Std. No. 245	Tolerances for Aluminum Alloy and Magnesium Alloy Wrought Parts

3. REQUIREMENTS

3.1 Material. The material shall be furnished as composite flat sheet formed by diffusion bonding of boron filaments and aluminum alloy in such manner the filaments are solidly embedded in an aluminum alloy matrix.

3.1.1 Boron filament. The boron filaments shall be of 0.0056-inch diameter and shall possess an average tensile strength of 500 ksi.

3.1.2 Aluminum alloy. The aluminum alloy shall be 6061-F unless otherwise specified on the contract or purchase order and shall meet the requirements of Alcoa specifications for 6061 material.

3.1.3 Filament alignment. All filaments comprising a single ply shall be laid parallel one to another within one degree of the long axis of the ply.

3.1.4 Plies. The longitudinal direction of each ply used in compositing the material as related to the long axis of the sheet shall be as specified on the contract or purchase order.

3.2 Physical properties. The material shall meet the requirements of Table I.

Table I
Physical Properties (at room temperature)
Minimum Values

Material Type and Grade	Tensile Strength Long.	Tensile Strength Trans.	Modulus of Elasticity
Type I			
Grade A, psi	170,000	12,000	32×10^6
Type II (0-90° CP)			
Grade B, psi	70,000	70,000	19×10^6

3.3 Dimensions. The composite sheet shall be furnished in the thickness and size as specified on the contract or purchase order. Material may be furnished in a rough untrimmed condition provided that sufficient materials allowed to meet the specified size after trimming.

3.3.1 Tolerances. Tolerance on the thickness of composite sheet shall be ±10%.

3.4 Finish. Unless otherwise specified on the contract or purchase order the material shall be furnished in the mill finish.

3.5 Surface defects. The surface shall be free from cracks, scratches, folds, wrinkles, laps, indentions, edge delaminations, foreign objects, or other defects which would adversely affect the properties or serviceability of the material.

3.6 Internal defects. The material shall be free from voids, delaminations, stray filaments, broken filaments, filament and ply misalignment, and foreign matter. Occasional individual broken filaments are not cause for rejection. Concentrated groups of broken filaments shall be cause for rejecting.

3.7 Boron filament per cent by volume. Material percentage of boron filament content by volume shall be in accordance to grade as specified herein.

3.8 Product markings. The material shall be legibly identified with the following information.

a. 72C0191and applicable dash number.
b. Purchase order number.
c. Manufacturer's name.
d. Alloy and temper as applicable.
e. Size of material.
f. Lot number.

The marking material shall be such as to resist obliteration during normal handling and shall be removable by normal cleaning methods; however ghost images of the characters may remain. Markings shall appear at each end of the material.

3.9 Workmanship. The material shall be of uniform quality and condition, free from protruding filament ends and burrs.

4. QUALITY ASSURANCE PROVISIONS

4.1 Responsibility for inspection and test. Unless otherwise specified in the contract or purchase order, the seller is responsible for the performance of all inspection and test requirements as specified herein.

4.2 Inspection records. Inspection records of examinations and tests shall be kept complete and available to Convair. These records shall contain all data necessary to determine compliance with the requirements of this specification.

4.3 Classification of examinations and tests. The examinations and tests of the material shall be classified as follows:

a. Qualification verification
b. Acceptance verification
c. Receiving inspection

4.3.1 Qualification verification. Qualification verification shall consist of all the examinations and tests specified herein.

4.3.2 Acceptance verification. Acceptance verification shall be performed on representative samples of each lot of material, and shall consist of the following:

a. Examination of product
b. Tensile strength
c. Modulus of elasticity

4.4 Test conditions. Test conditions shall be as specified in 4.6.

4.5 Test methods.

4.5.1 Examination of product. The material shall be examined to verify that the markings, size, surface, and workmanship conform to the requirements of this specification.

4.5.2 Tensile strength and modulus of elasticity. Compliance with the requirements of 3.2, Table I shall be determined in accordance with Federal Test Method Standard No. 151a, Method 211.1.

4.5.3 Internal defects. Compliance with the requirements of 3.6 shall be determined by inspections, tests and methods agreed upon by Convair and Vendor.

4.5.4 Boron filament per cent by volume. Compliance with the requirements of 3.7 shall be determined by an inspection method agreed upon by Convair and Vendor.

5. PREPARATION FOR DELIVERY

5.1 Preservation and packaging. Preservation and packaging of all material furnished under this specification shall be sufficient to afford adequate protection against corrosion and physical damage during handling, shipping, and storage. Each package or container shall contain only material from the same lot.

5.2 Packing. The material shall be packaged as specified in 5.1 and packed in a manner which will ensure acceptance by common carrier at lowest rates and will ensure protection against damage during shipment.

5.3 Marking for shipment. Each shipping container shall be identified with label, tag, or marking which includes the following data.

a. 72C0191 and applicable dash number.
b. Purchase order number.
c. Manufacturer's name.
d. Material description.
e. Quantity and unit size.
f. Lot number.
g. Precautionary, handling, and storing warnings, as applicable.

6. NOTES

6.1 Intended use. The material covered by this specification is intended to be used in the manufacture of structural components when the composite properties of high modulus filament and aluminum alloy matrix are desirable. Use is not restricted to this application.

6.2 Ordering information. The following information should be included on the purchase order.

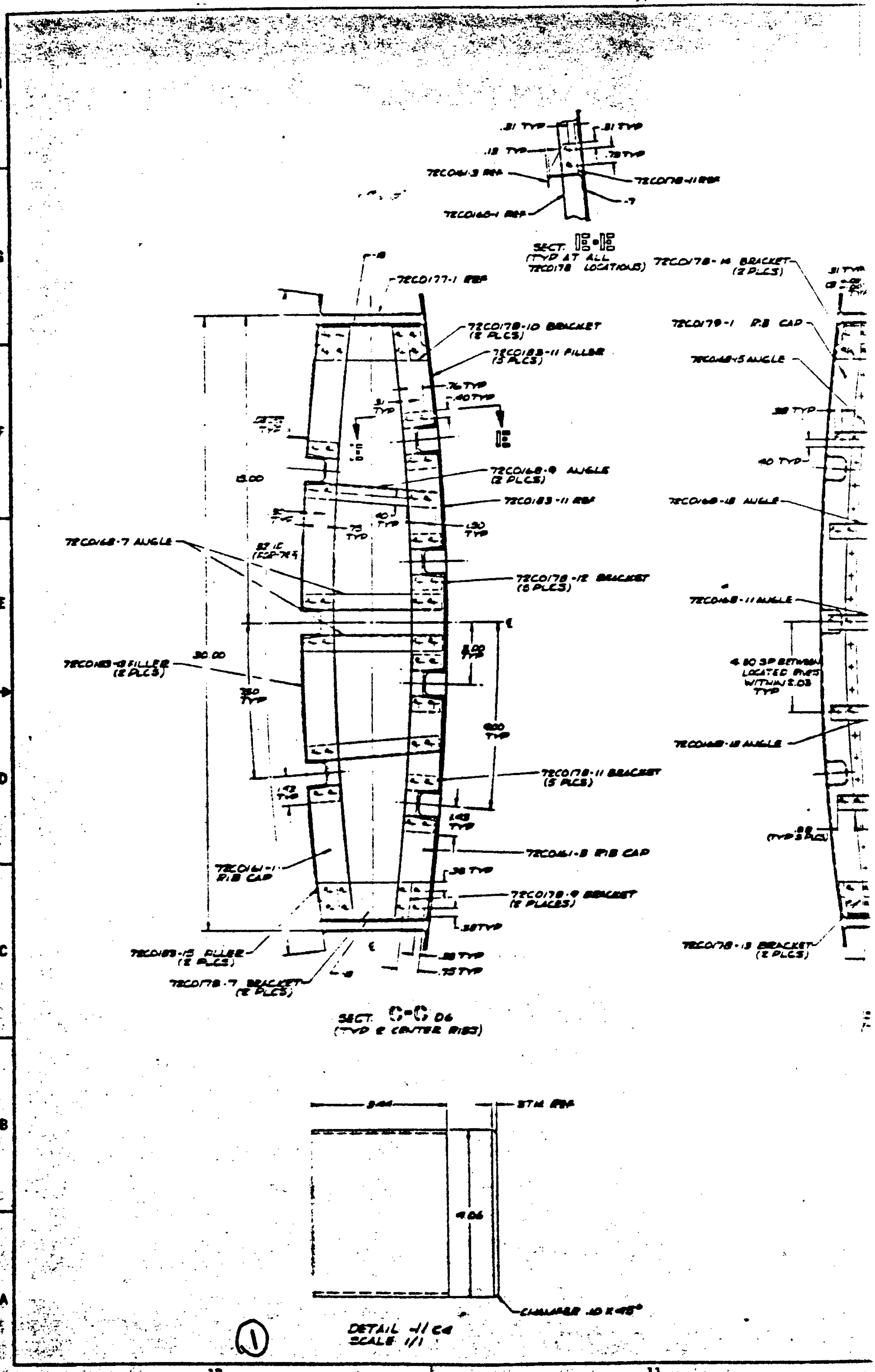
a. Number, title, and date of this specification.
b. Lay of plies.
c. Size and thickness of composite sheet.
d. Quantity.
e. Material classification identification.

6.3 Material classification identification. The classification identification numbers for the material specified herein shall consist of the number of this specification and the applicable dash number as given below:

Type	Grade	Convair Designation
I	A	72C0191-1
II	B	72C0191-2

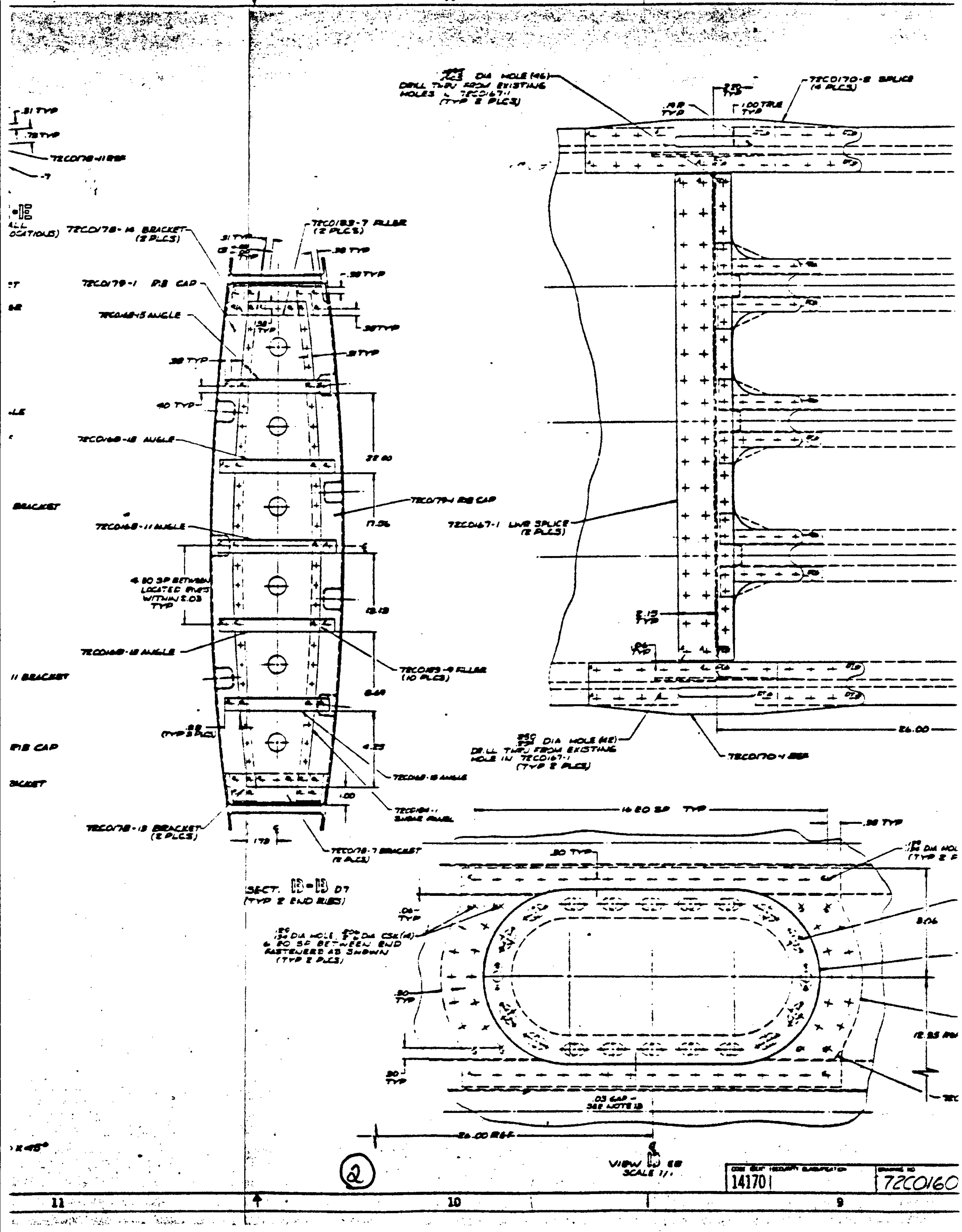
APPENDIX B

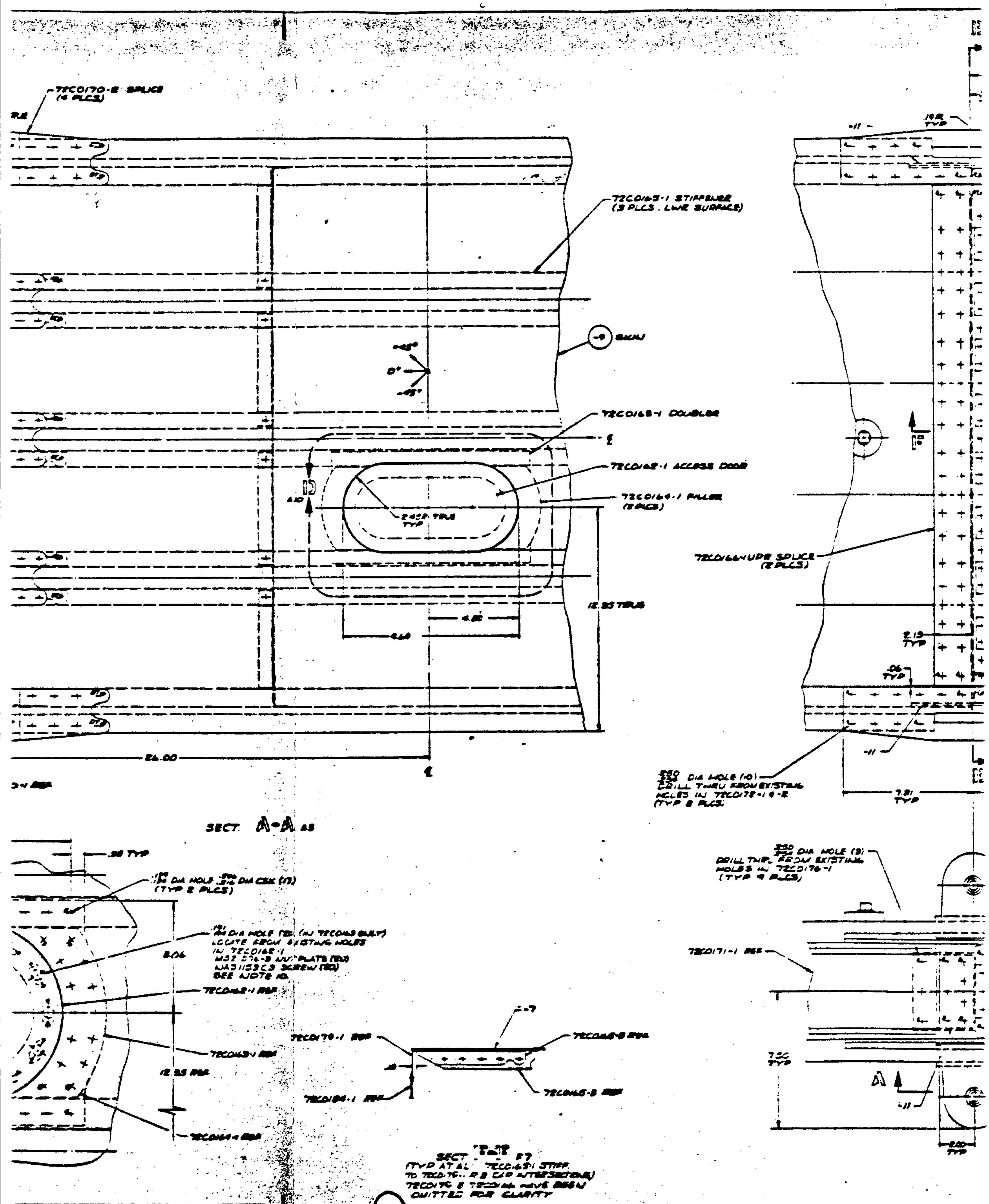
WING BOX ASSEMBLY DRAWINGS



SECT. E-E
(TYP AT ALL 72C0178 LOCATIONS)
72C0177-1 REF
72C0178-10 BRACKET (2 PLCS)
72C0183-11 FILLER (3 PLCS)
72C0168-9 ANGLE (2 PLCS)
72C0183-11 REF
72C0168-7 ANGLE
72C0178-12 BRACKET (5 PLCS)
72C0183-13 FILLER (2 PLCS)
72C0178-11 BRACKET (5 PLCS)
72C0161-3 RIB CAP
72C0161-1 RIB CAP
72C0178-9 BRACKET (2 PLACES)
72C0183-15 FILLER (2 PLCS)
72C0178-7 BRACKET (2 PLCS)
SECT. C-C D6
(TYP 2 CENTER RIBS)
72C0178-14 BRACKET (2 PLCS)
72C0179-1 RIB CAP
72C0168-11 ANGLE
72C0178-13 BRACKET (2 PLCS)
30.00
13.00
STA REF
4.06
CHAMFER .10 X 45°
DETAIL -1/C4
SCALE 1/1

12
11

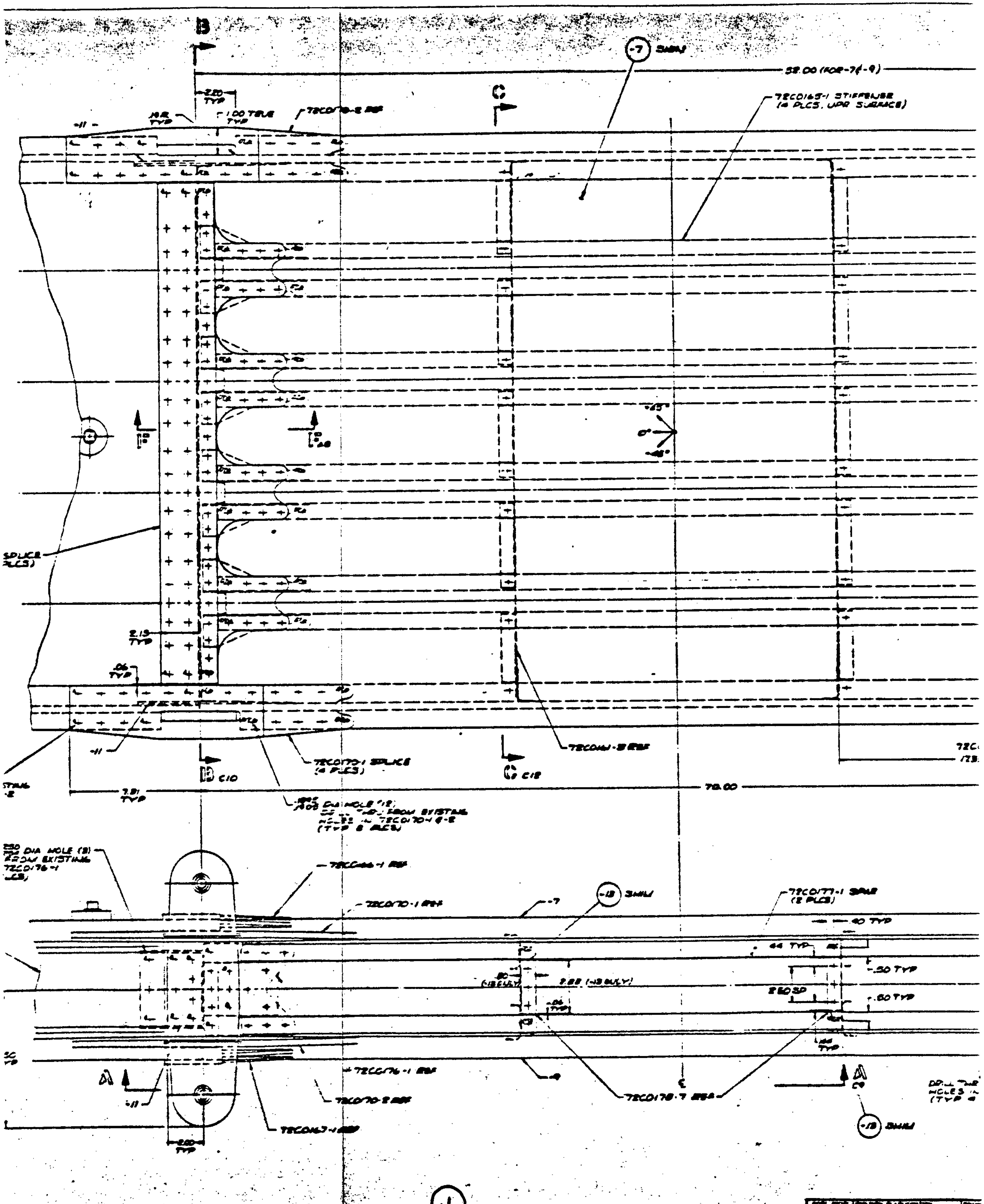

72C0170-2 SPLICE (4 PLCS)
DRILL THRU FROM EXISTING HOLES IN 72C0167-1 (TYP 2 PLCS)
72C0178-14 BRACKET (2 PLCS)
72C0183-7 FILLER (2 PLCS)
72C0179-1 RIB CAP
72C0179-1 RIB CAP
72C0168-12 ANGLE
72C0168-11 ANGLE
72C0168-13 ANGLE
72C0183-9 FILLER (10 PLCS)
72C0168-15 ANGLE
72C0184-1 SHEAR PANEL
72C0178-13 BRACKET (2 PLCS)
72C0178-7 BRACKET (2 PLCS)
4.80 SP BETWEEN LOCATED RIVETS WITHIN 2.03 TYP
72C0167-1 LWR SPLICE (2 PLCS)
DIA HOLE (42) DRILL THRU FROM EXISTING HOLE IN 72C0167-1 (TYP 2 PLCS)
72C0170-4 SPLICE
16.80 SP TYP
24.00
.03 GAP - SEE NOTE 13
24.00 REF
SECT. B-B D7 (TYP 2 END RIBS)
VIEW B 8B SCALE 1/1
14170
72C0160
11
10
9



DRAWING NO	SH
72C0160	1

3

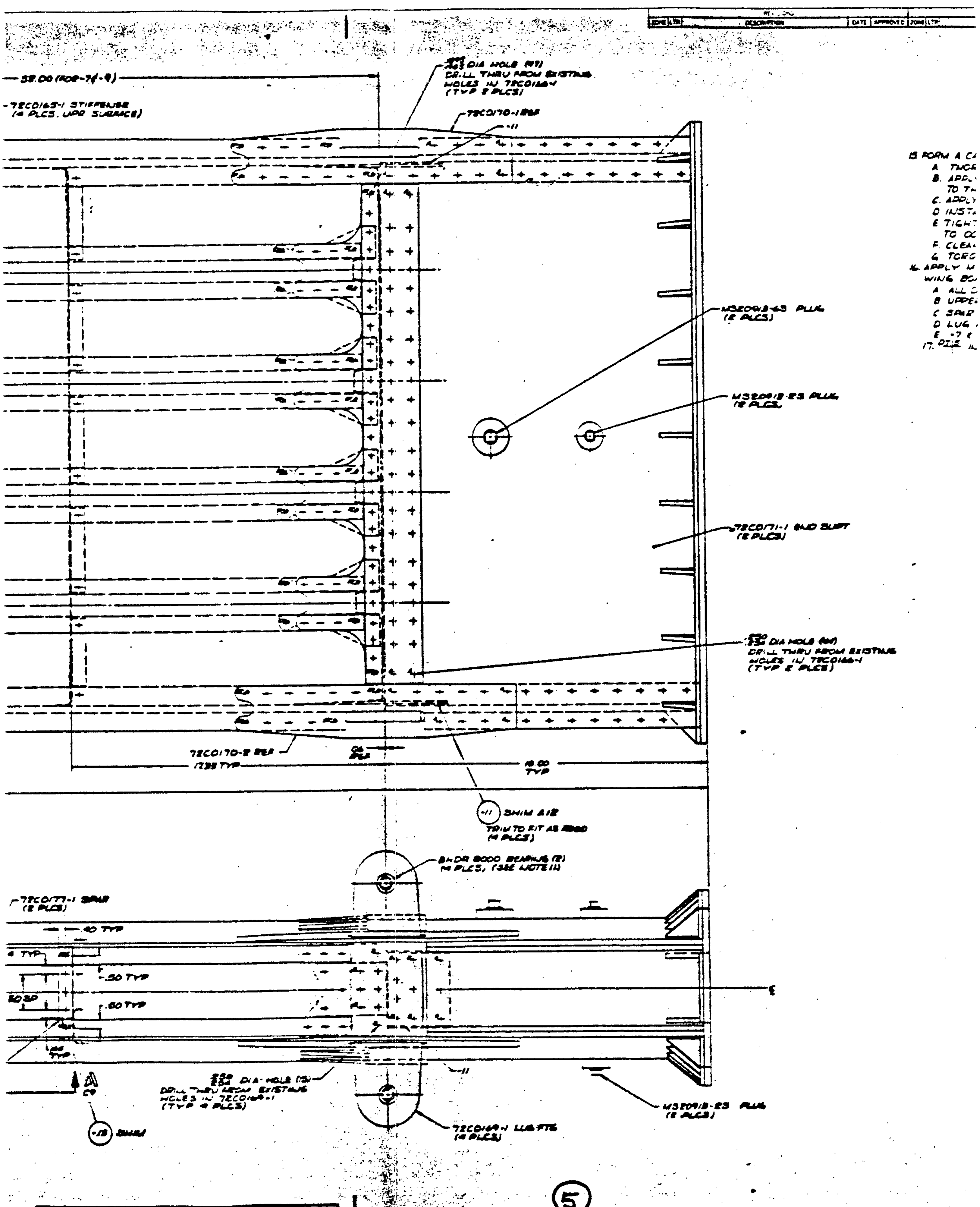
9 8 7



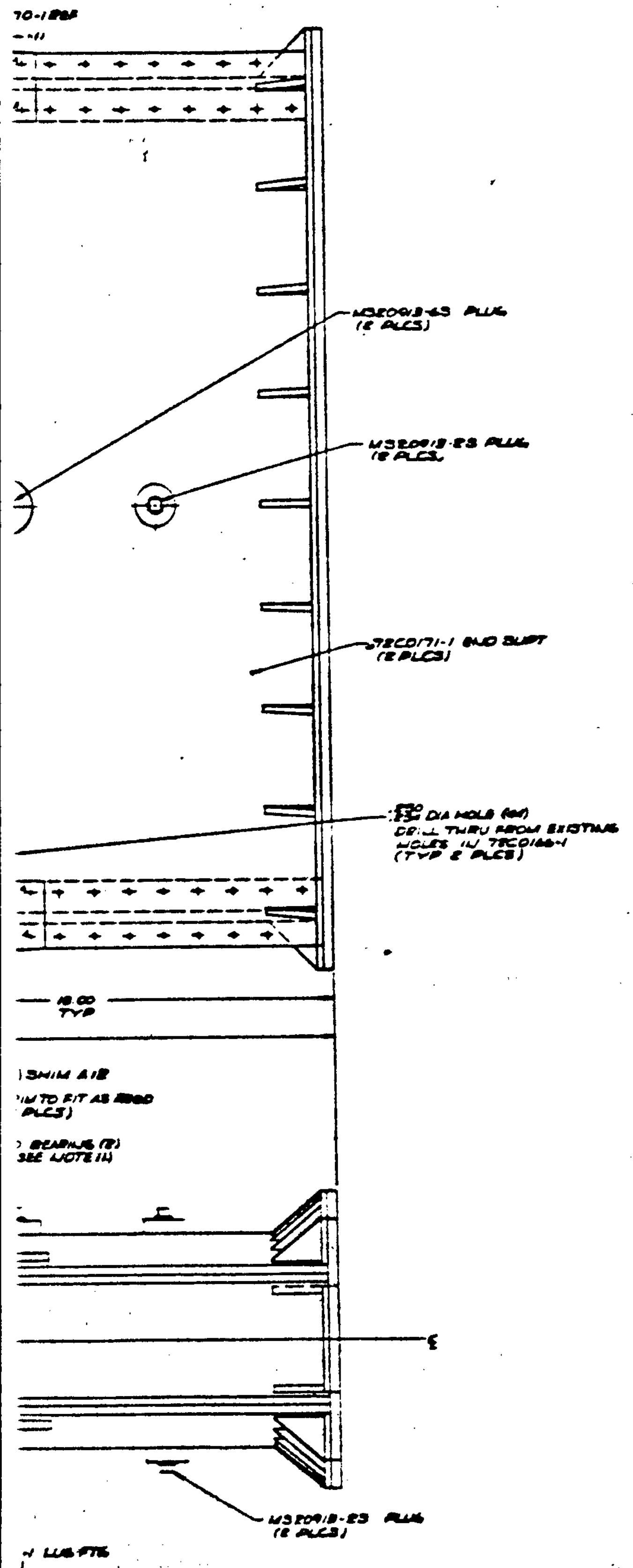
-7 SHIM
58.00 (FOR -7 & -9)
72C0165-1 STIFFENER (4 PLCS, UPR SURFACE)
.220 TYP
1.00 TRUE TYP
72C0170-2 REF
SPLICE
2.15 TYP
.06 TYP
-11
72C0170-1 SPLICE (4 PLCS)
72C0161-3 REF
78.00
7.81 TYP
72C0166-1 REF
72C0170-1 REF
-13 SHIM
72C0177-1 SPAR (2 PLCS)
.40 TYP
.44 TYP
.50 TYP
.60 TYP
2.80 SP
2.88 (-13 ONLY)
72C0176-1 REF
72C0170-2 REF
72C0167-1 REF
72C0178-7 REF
.200 TYP
-9
-13 SHIM

14170

4



CODE IDENT	SECURITY CLASSIFICATION	DRAWING NO.	SH
14170		72C0160	7



15. FORM A CAST IN PLACE SEAL FOR 72C0162-1 DOOR USING FOLLOWING PROCEDURE:
   A. THOROUGHLY CLEAN DOOR LANDING & DOOR.
   B. APPLY AN EVEN COATING OF MIL-S-8802-C-20 SEALANT, .03-.05 THICK TO THE DOOR LANDING
   C. APPLY A THIN COAT OF VASELINE TO MATING SURFACE AS A PARTING AGENT.
   D. INSTALL DOOR USING FASTENERS WIPED WITH VASELINE.
   E. TIGHTEN FASTENERS IN AN EVEN PATTERN UNTIL SEALANT BEGINS TO OOZE FROM JOINT
   F. CLEAN OFF EXCESS SEALANT & ALLOW TO CURE.
   G. TORQUE FASTENERS TO NORMAL VALUE.
16. APPLY MIL-S-8802C-20 SEALANT TO THE FAYING SURFACES OF THE FOLLOWING WING BOX COMPONENTS DURING ASSEMBLY.
   A. ALL DETAILS BOLTED TO 72C0171 END SUPPORT.
   B. UPPER & LWR FINGERED SPLICES, 72C0166 & 72C0167.
   C. SPAR WEB SPLICE 72C0176 TO 72C0171 END SUPPORT.
   D. LUG FITTING 72C0169 TO SPAR WEB.
   E. -7 & -9 SKINS TO SPAR CAPS
17. DIS INDICATES NAS1463 PIN & NAS1080-5 COLLAR

NOTES:
1 ... IND
2 ... IND
3 ... IND
4. ... IND
5. USE AN ... AS REQ
6. IMPRESS
7. BREAK
8. ALL DIM
9. INSTALL ... PER
10. INSTALL ... WITH
11. STAKE
12. ... IND
13. TRIM FL
14. ... IND

QTY	PART
8	BHD.
AR	NAS
AR	NAS
AR	NAS
AR	NAS
AR	NAS
AR	NAS
AR	MS
20	NAS
20	MS2
AR	MS2
2	MS2
4	MS2
AR	AN
AR	NAS
4	72C
2	72C
4	72C
4	72C
10	72C
20	72C
4	72C
4	72C
4	72C
4	72C
2	72C
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4	72C
4	72C
2	72C
2	72C
2	72C
2	72C
2	72C
2	72C
4	
4	
1	
REQUIREMENTS PER ASSEMBLY / INSTALLATION	UNIT OF MEAS / ITEM OF PROC NC / CODE IDENT NO

UNLESS OTHERWISE SPECIFIED
ALL MACHINED SURFACES √

6

4 3 2

ZONE	LTR	DESCRIPTION	DATE	APPROVED
	A	ADDED SECT F-F. IN SECT. B-B ADDED REVISED 72C0179-1 E.B CAP. IN SECT A-A & MAIN VIEW ADDED REVISED 72C0163-1 [illegible]		
	B	ADDED -9 SKIN & DTS(2) (72C0...)		

62-1 DOOR USING FOLLOWING PROCEDURE:
ING & DOOR
S-8802-C-20 SEALANT, .03-.05 THICK
MATING SURFACE AS A PARTING AGENT.
WIPED WITH VASELINE.
PATTERN UNTIL SEALANT BEGINS
LLOW TO CURE.
VALUE.
FAYING SURFACES OF THE FOLLOWING
EMBLY.
END SUPPORT.
72C0166 & 72C0167.
C0171 END SUPPORT.
...
S1080-5 COLLAR

NOTES:

1. INDICATES. NAS1466 PIN & NAS1080-6 COLLAR.
2. INDICATES. NAS1004-9A BOLT & MS21042-4 NUT.
3. INDICATES. NAS1004-18A BOLT & MS21042-4 NUT.
4. INDICATES. FASTENER INSTALLED ON OTHER DWG.
5. USE AN960-416 WASHERS WITH NAS1004 BOLTS & MS21042-4 NUTS AS REQUIRED TO INSURE INTERFACE CLAMP UP.
6. IMPRESSION MARKING OF PART NOT PERMITTED.
7. BREAK SHARP EDGES, .015
8. ALL DIMENSIONS ARE IN PLANE OF PAPER EXCEPT AS NOTED.
9. INSTALL ALL FASTENERS IN WET, UNTHINNED ZINC CHROMATE PRIMER PER 0-00301
10. INSTALL MS21076-3 NUTPLATES ON INBD SIDE OF 72C0163-1 DOUBLER WITH MS20426AD3 RIVETS.
11. STAKE BEARINGS PER 0-70011.
12. INDICATES. NAS1399D4 RIVET.
13. TRIM FLANGE OF 72C0165-1 STIFFENER TO WIDTH OF ACCESS DOOR OPENING, AS SHOWN.
14. INDICATES. NAS1398D4 RIVET.

REQUIREMENTS PER ASSEMBLY / INSTALLATION	UNIT OF MEAS	ITEM OR FIND NO	CODE IDENT NO	PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION	ZONE	NOTE	STOCK SIZE, MATERIAL SPECIFICATION, TYPE, FINISH, ETC.
8				BHCR 8000	BEARING	C 4		SOUTHWEST PRODUCTS CO
48				NAS1465	PIN			
48				NAS1466	PIN			
48				NAS1399D4	RIVET			
46				NAS1398D4	RIVET			
42				NAS1080-6	COLLAR			
[illegible]				NAS1004-21	BOLT			
48				NAS1004-9A	BOLT			
20				NAS[illegible]	SCREW	B 9		
20				MS21076-3	NUTPLATE	B 9		
42				MS21042-4	NUT			
2				MS20392-[illegible]	[illegible]	F 8		
4				MS20613-[illegible]	[illegible]	[illegible]		
42				AN960-416	WASHER		5	SEE NOTE 5
42				NAS1080-5	COLLAR			
4				72C0170-2	SPLICE	H 9		
2				72C0184-1	SHEAR PANEL	C 10		
4				72C0183-13	FILLER	C 11		
4				72C0183-12		E 12		
10				72C0183-11		F 11		
20				72C0183-9		D 10		
4				72C0183-7	FILLER	G 10		
4				72C0179-1	E B CAP	E 10		
4				72C0178-14	BRACKET	G 11		
4				72C0178-13		G 11		
2				72C0178-12		F 11		
2				72C0178-11		G 11		
4				72C0178-10		F 11		
4				72C0178-5		G 11		
8				72C0178-1	BRACKET	C 12		
2				72C0177-[illegible]	[illegible]	C 5		
2				72C0171-1	END SUPT	E 8		
4				72C0170-1	SPLICE	C 6		
4				72C0169-1	FITTING	A 6		
4				72C0168-15	ANGLE BRACKET	[illegible]		
4				72C0168-[illegible]		F [illegible]		
2				72C0168-11		E [illegible]		
4				72C0168-5		[illegible]		
4				72C0168-1	ANGLE BRACKET	[illegible]		
2				72C0167-1	[illegible] SPLICE	[illegible]		
2				72C0166-1	UPR SPLICE	[illegible]		
-				72C0165-1	STIFFENER	D 8		
2				72C0164-1	FILLER	E 8		
				72C0163-1	DOUBLER	F 8		
				72C0162-1	ACCESS DOOR	F 8		
2				72C0161-2	F B CAP	[illegible]		
2				72C0161-1	F B CAP	E 2		
4				-12	SHIM	[illegible]	W	[illegible] ALUMINUM
4				-11	SHIM	[illegible]	W	[illegible] ASTM-A7 STEEL
1				-9	SKIN	F 8	W	[illegible] BORON ALUMINUM
1				-7	SKIN	[illegible] 6	W	[illegible] BORON ALUMINUM
-				-1	ASSY		B	END ITEM

PARTS LIST

UNLESS OTHERWISE SPECIFIED
ALL MACHINED SURFACES √

SEE 0-70014 FOR ABBREVIATION CODE AND SYMBOL REQUIREMENTS

SEE 0-70000 FOR STANDARD MANUFACTURING AND INSTALLATION REQUIREMENTS

ASSOCIATED LIST REQUIRED

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES
TOLERANCES

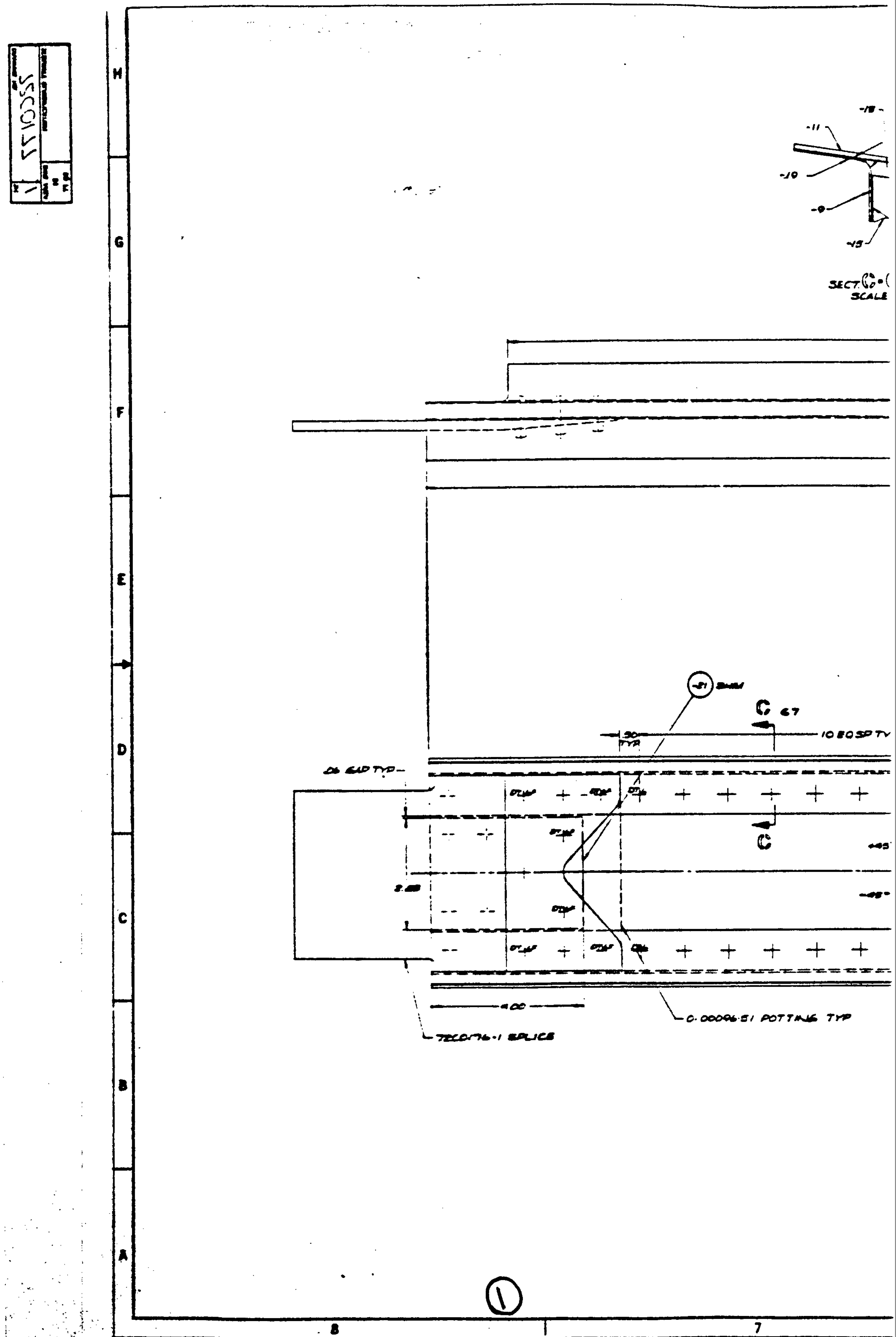
CONVAIR DIVISION OF GENERAL DYNAMICS
SAN DIEGO CALIFORNIA

BORON-ALUMINUM WING BOX.
TEST SPECIMEN.
ASSY OF

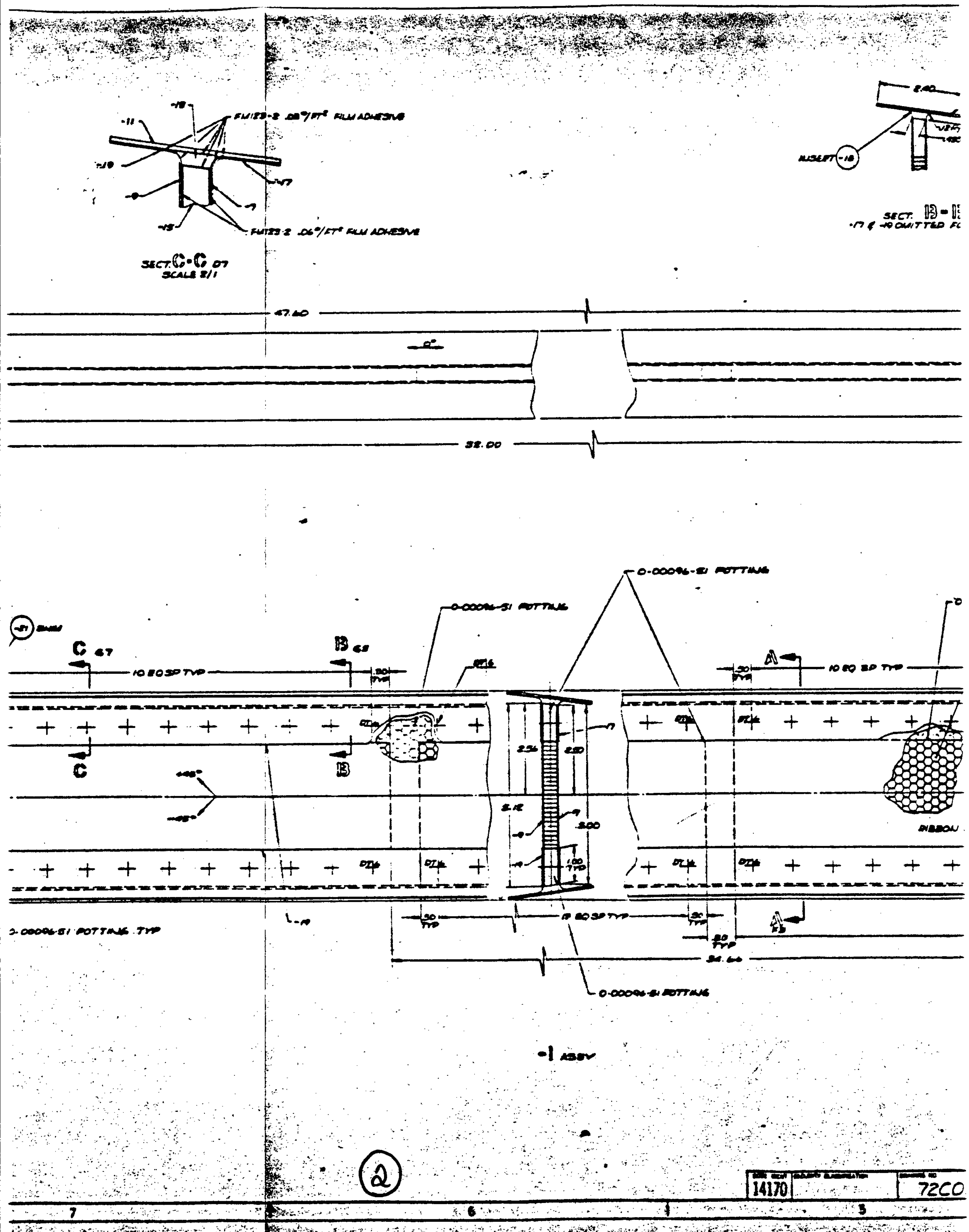
SIZE	CODE IDENT NO	DRAWING NO
J	14170	72C0160

(7)

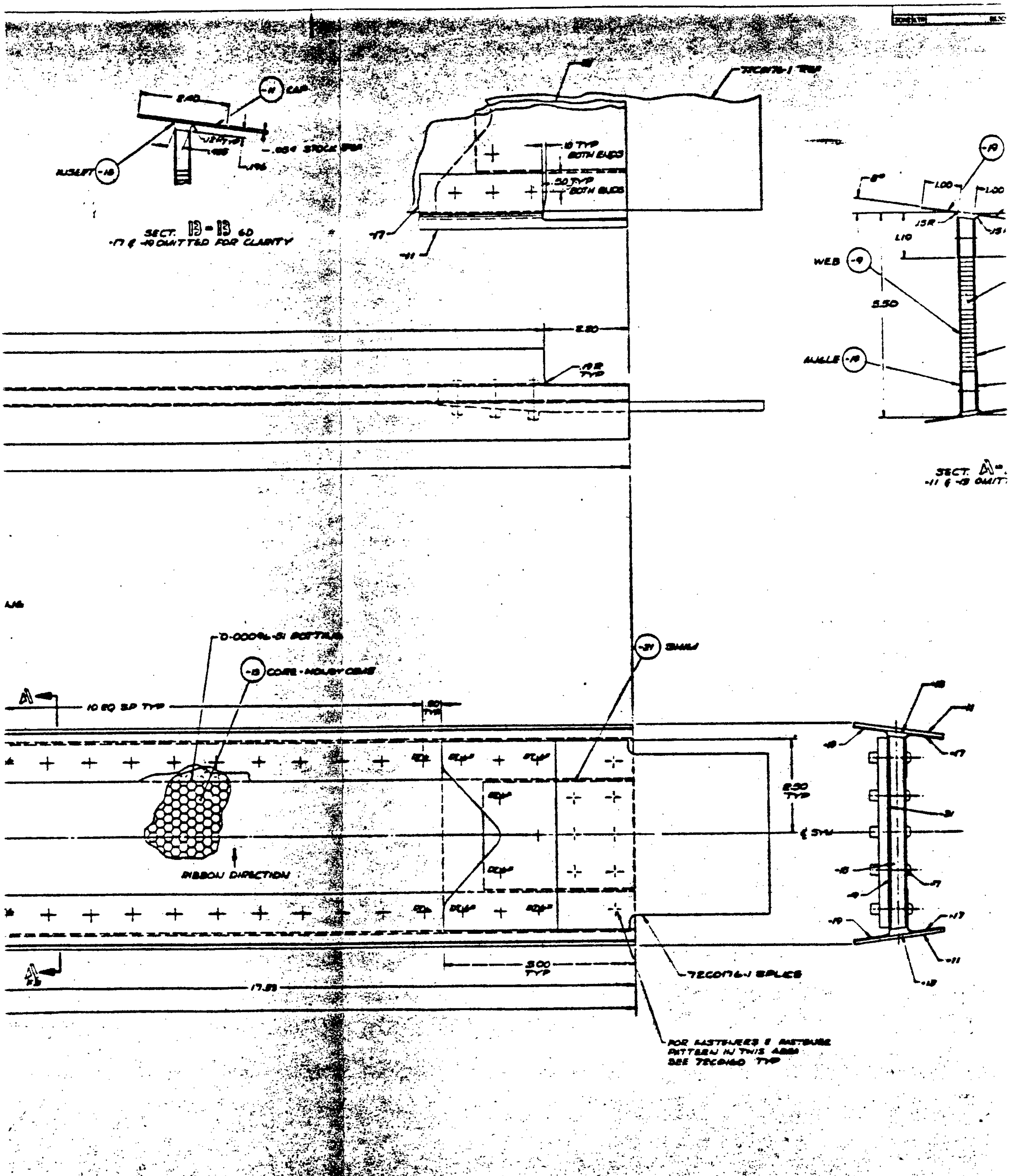
2



H
G
F
E
D
C
B
A
-11
-10
-9
-15
SCALE
10 EQ SP TY
.30 TYP
.06 GAP TYP
2.88
4.00
SPLICE
POTTING TYP
8
7



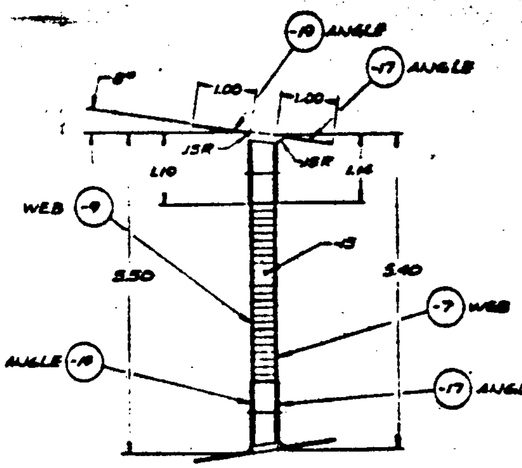
FM123-2 .08#/FT² FILM ADHESIVE
FM123-2 .06#/FT² FILM ADHESIVE
SECT. C-C D7
SCALE 2/1
SECT. B-B
47.60
32.00
0-00096-S1 POTTING
0-00096-S1 POTTING
0-00096-S1 POTTING TYP
0-00096-S1 POTTING
10 EQ SP TYP
2.56
2.50
3.00
1.00 TYP
.50 TYP
34.66
RIBBON
-1 ASSY
2
14170
72C0



(3)

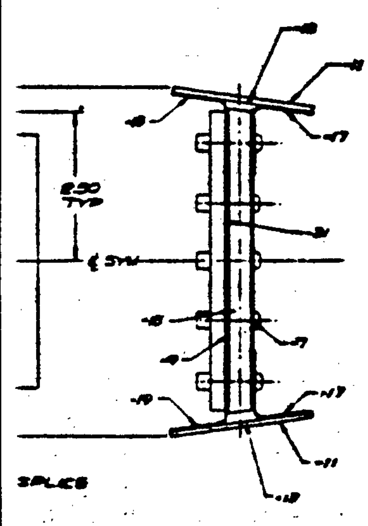
14170		72C0177	1

5 4 3



SECT. A-A
-11 & -13 OMITTED FOR CLARITY

NOTES:
1. IMPRESSION MARKING OF PARTS NOT PERMITTED
2. BREAK SHARP EDGES .015.
3. DYE PENETRANT INSPECT -7 & -9 DETAILS. NO C[illegible]
4. [illegible] NAS [illegible] PIN & NAS1080-6 COLLAR.
5. INSTALL ALL FASTENERS IN WET UNTHINNED ZINC [illegible]
6. APPLY MIL-S-8802C-20 SEALANT TO FAYING SURF[illegible] SPLICE TO SPAR WEB.



SPLICE

& FASTENER
3 AREA
TYP

QTY REQD	PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION	MATERIAL
80	[illegible]	PIN	
80	NAS1080 [illegible]	COLLAR	
AR	[illegible]	FILM ADHESIVE	
2	[illegible]	[illegible]	
2	-2	SHIM	[illegible]
2	-3	ANGLE	[illegible]
2	-[illegible]	ANGLE	[illegible]
1	-2	[illegible]	[illegible]
2	-3	[illegible]	[illegible]
2	[illegible]	[illegible]	[illegible]
1	[illegible]	WEB	[illegible]
1	[illegible]	WEB	[illegible]
[illegible]	[illegible]	SPLICE ASSY	[illegible]

PARTS LIST

UNLESS OTHERWISE SPECIFIED
DIMENSIONS ARE IN INCHES

(4)

3 4 2

NOTES:

1. IMPRESSION MARKING OF PARTS NOT PERMITTED.
2. BREAK SHARP EDGES. .015.
3. DYE PENETRANT INSPECT -7 & -9 DETAILS, NO CRACKS ALLOWED.
4. [illegible] = NAS [illegible] PIN & NAS1080-6 COLLAR.
5. INSTALL ALL FASTENERS IN WET UNTHINNED ZINC CHROMATE PRIMER PER 0-00801.
6. APPLY MIL-S-8802C-20 SEALANT TO FAYING SURFACES OF 72C0176 SPLICE TO SPAR WEB.

QTY REQD	PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION	MATERIAL
[illegible]	[illegible]	PIN	
[illegible]	NAS1080-6	COLLAR	
AR	[illegible]	FILM ADHESIVE	
2	72C0176	SPLICE	
2	-2	SHIM	[illegible] TITANIUM
2	-5	ANGLE	[illegible] TITANIUM
2	-3	ANGLE	[illegible] TITANIUM
1	-2	CORE	[illegible] HONEYCOMB
2	-8	WEB	[illegible] BORON-ALUMINUM
2	[illegible]	CAP	[illegible] BORON-ALUMINUM
1	[illegible]	WEB	[illegible] BORON-ALUMINUM
1	[illegible]	WEB	[illegible] BORON-ALUMINUM
[illegible]	[illegible]	SPAR ASSY	[illegible] ASSY 72C0160 [illegible]

PARTS LIST

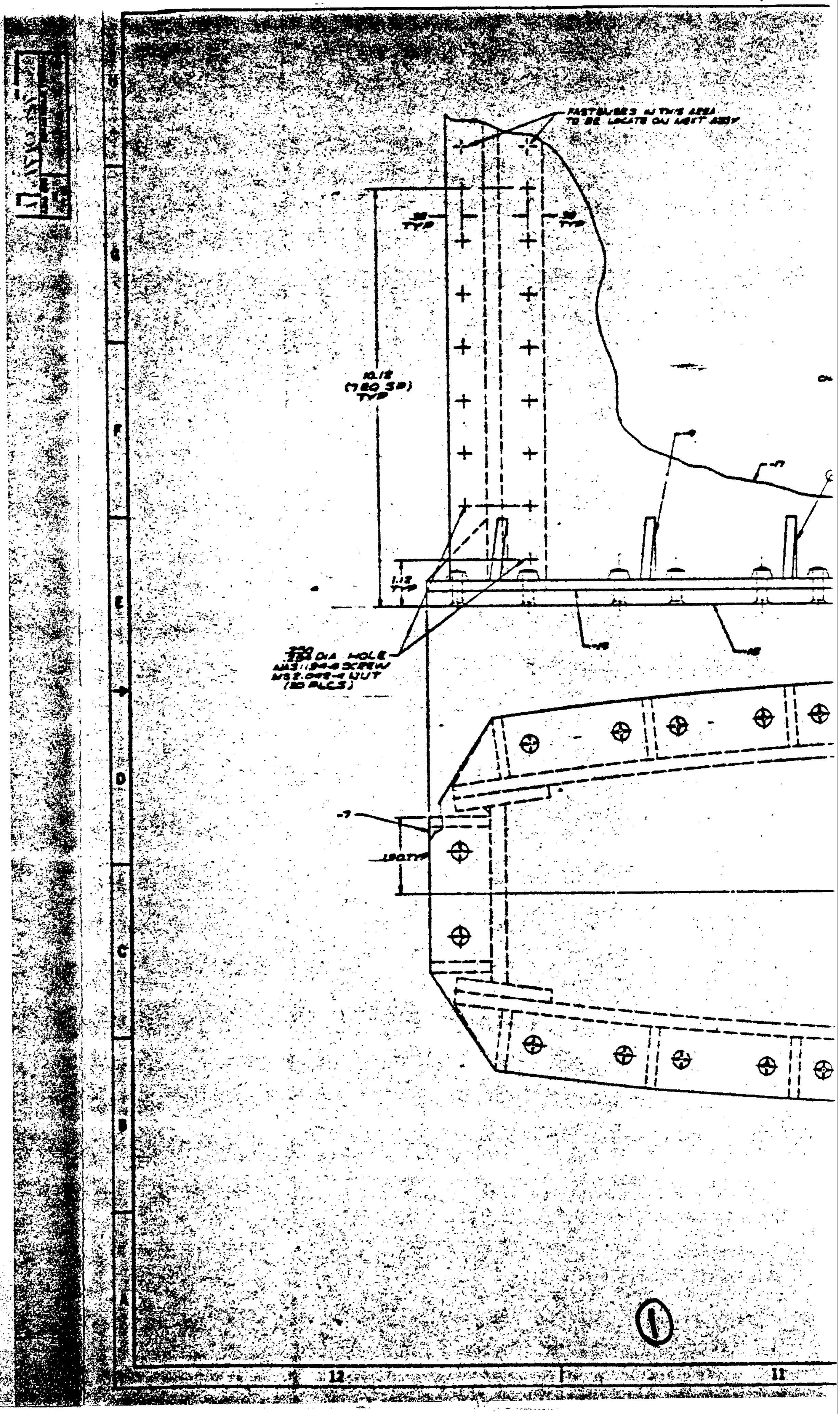
CONVAIR DIVISION OF GENERAL DYNAMICS
SAN DIEGO, CALIFORNIA

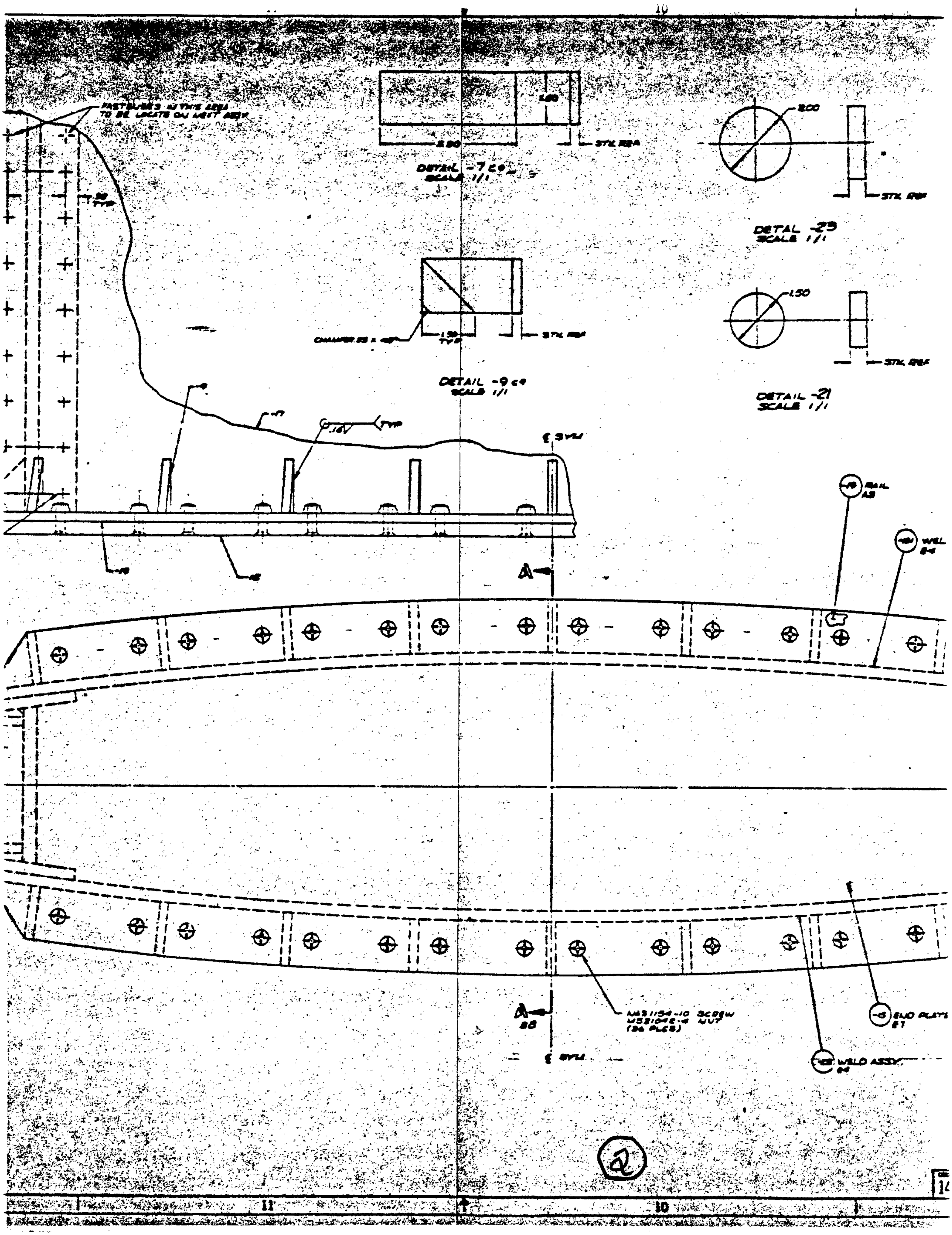
SPAR - ASSY OF,
BORON ALUMINUM WING BOX
TEST SPECIMEN

J 14170 72C0177

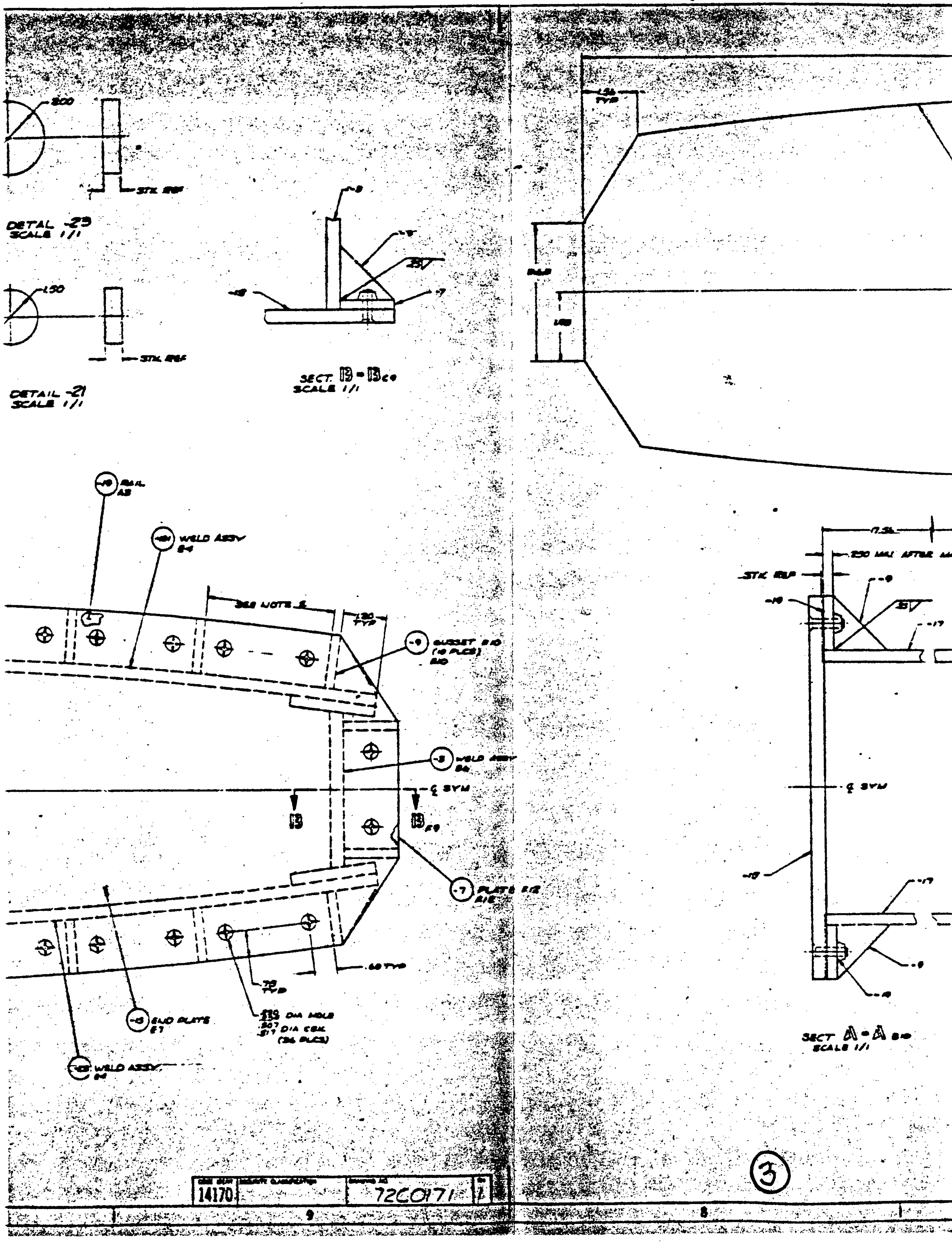
5

2

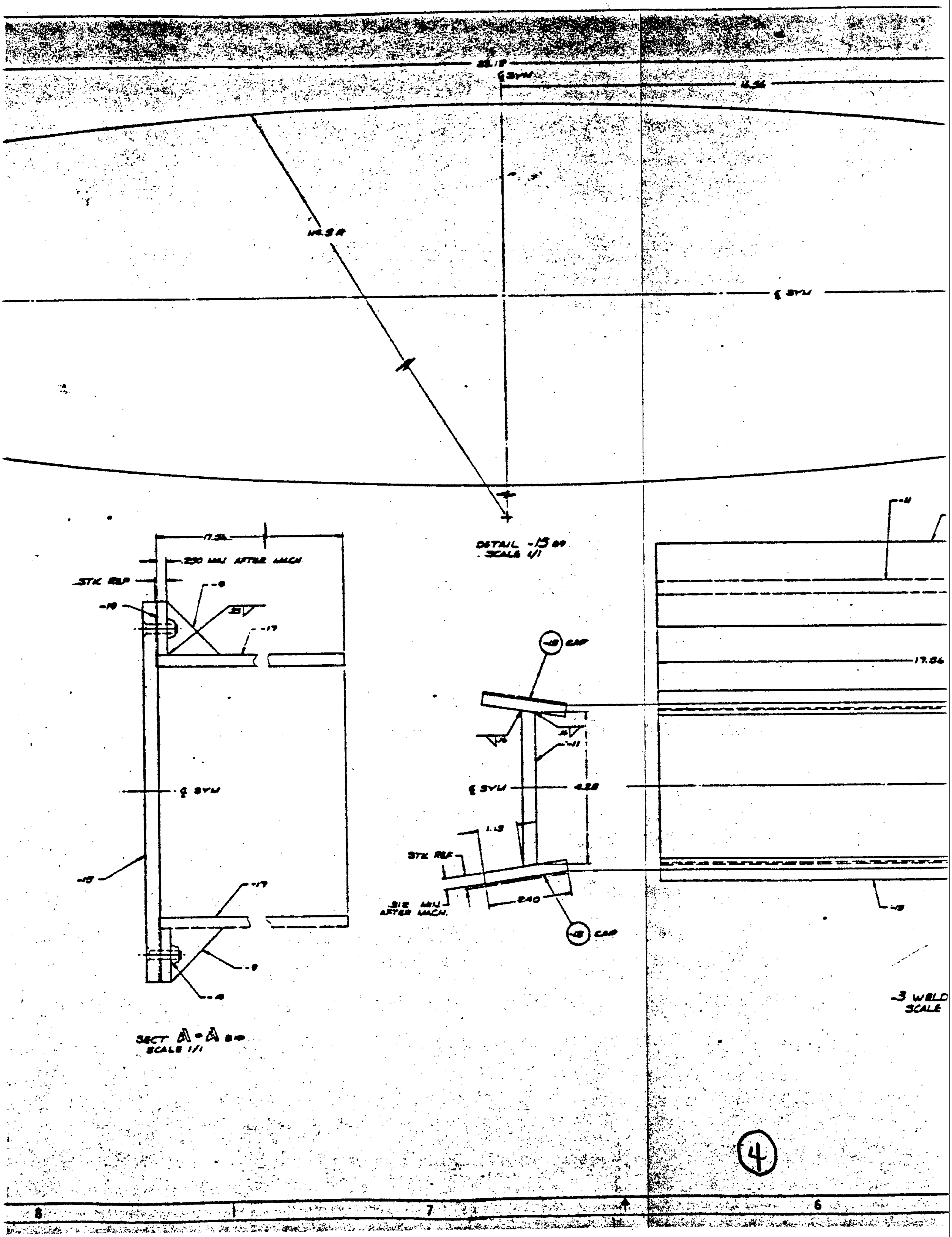

H
G
F
E
D
C
B
A
TYP
1.12 TYP
DIA HOLE
-7
-17
-18
12
11



STK REF
SCALE 1/1
DETAIL -23
SCALE 1/1
DETAIL -9
SCALE 1/1
DETAIL -21
SCALE 1/1
STK REF
TYP
WELD ASSY
SYM
11
10

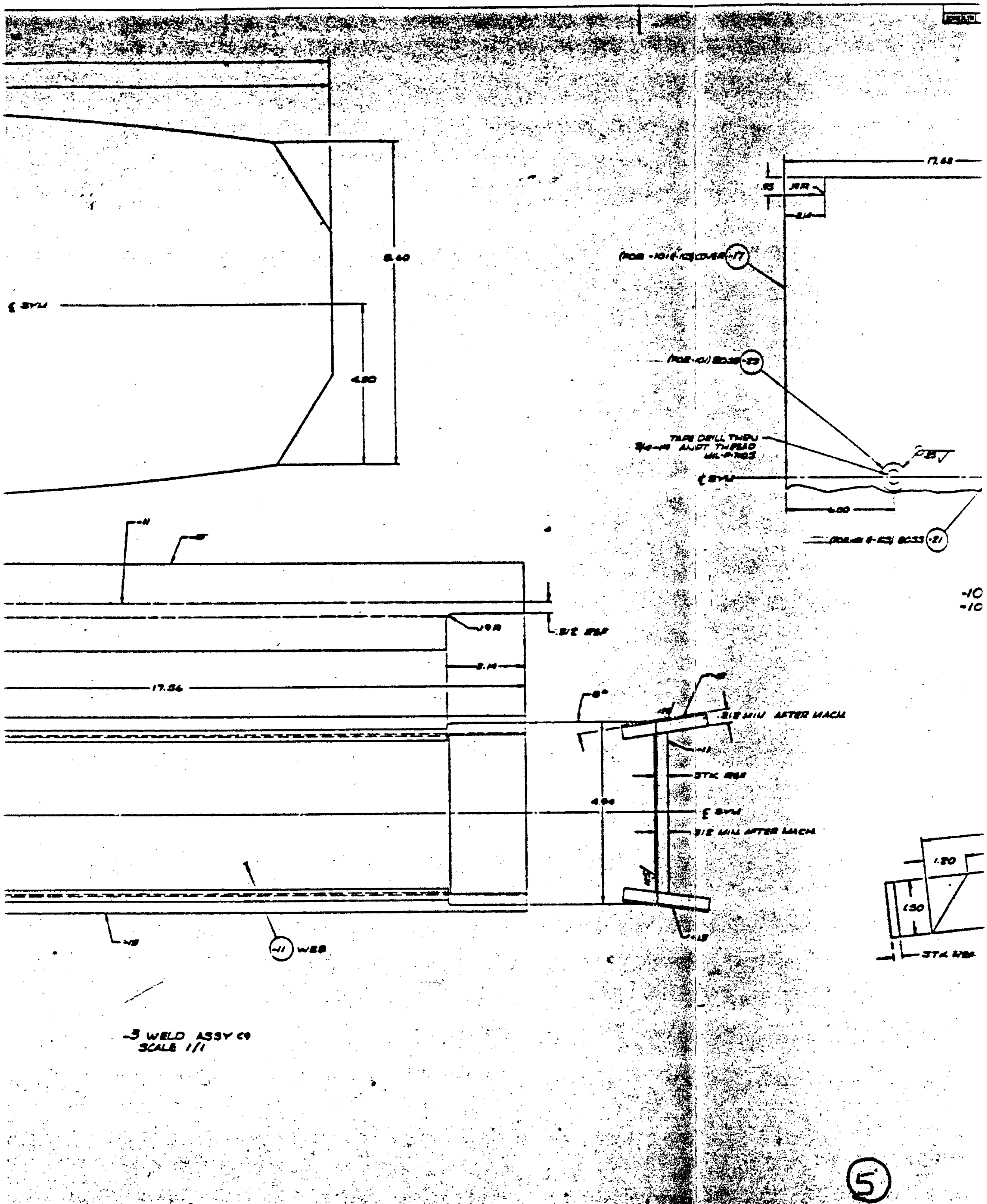


DETAIL -23
SCALE 1/1
DETAIL -21
SCALE 1/1
SECT. B-B
SCALE 1/1
STK REF
WELD ASSY
GUSSET
WELD ASSY
PLATE
¢ SYM
END PLATE
WELD ASSY
DIA HOLE
SECT A-A
SCALE 1/1
14170
72C0171
3



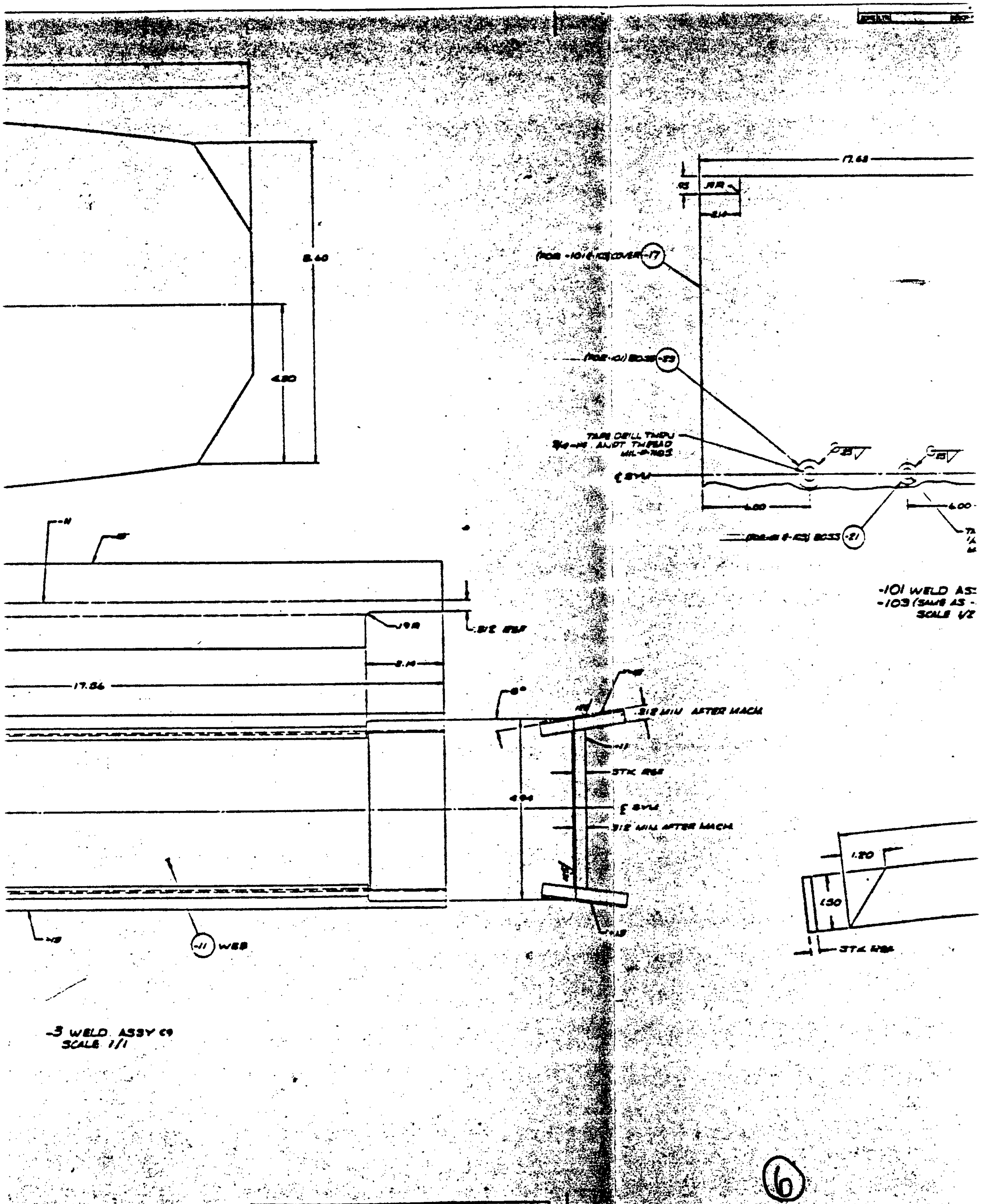
℄ SYM
DETAIL -13
SCALE 1/1
.250 MIN AFTER MACH.
STK REF
℄ SYM
SECT A-A
SCALE 1/1
-13 CAP
4.28
1.13
STK REF
.312 MIN. AFTER MACH.
.240
17.86
-3 WELD
SCALE

4



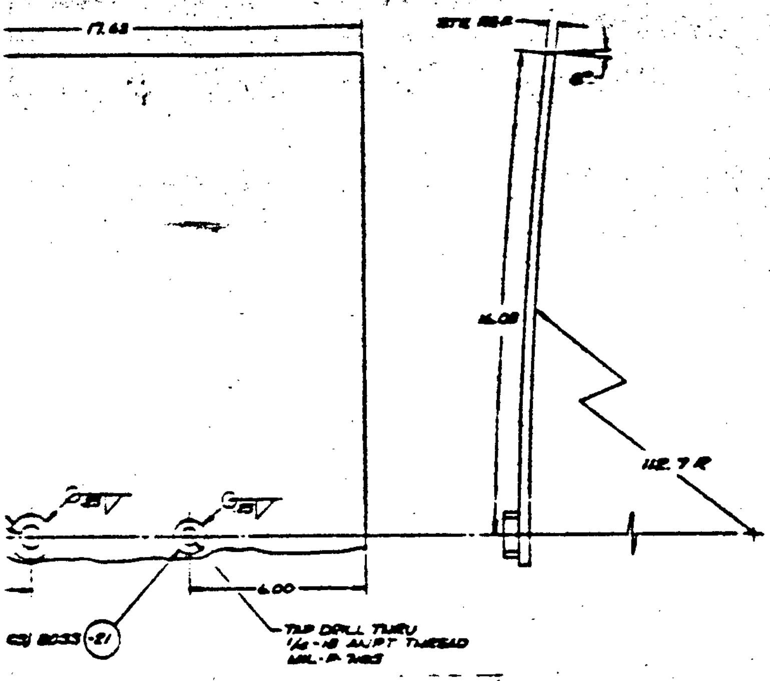
14170		72C0171	7

6 5 4

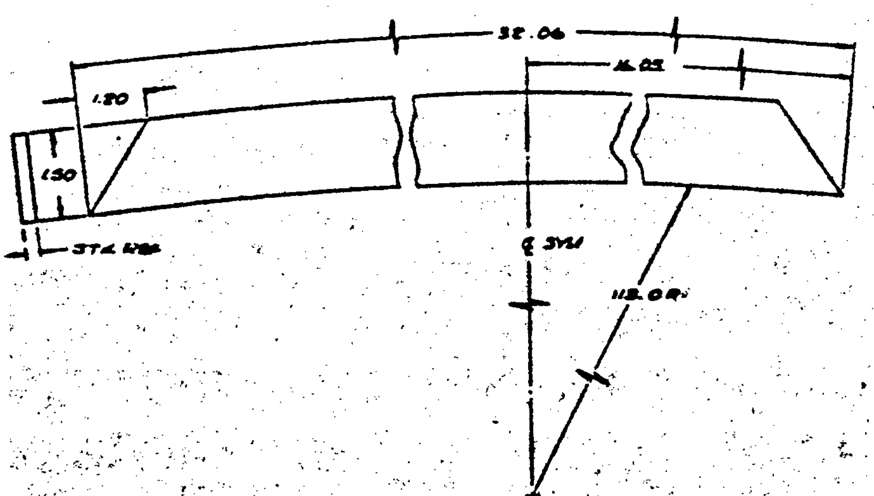


6

14170 72C0171 5



-101 WELD ASSY E9
-103 (SAME AS -101 EXCEPT AS NOTED) B10
SCALE 1/2



DETAIL -19 B10
SCALE 1/1

7

NOTES:
1. IMPRESSION M
2. BREAK SHARP
3. MINIMUM CLEAN
COMPONENTS
4. WELD PER MIL-
5. -9 GUSSETS TO
AS SHOWN
6. MAGNETIC INSP
7. ACID RESIN PRI
8. APPLY O-7525?
9. APPLY O-75C5
10. APPLY MIL-S-
11. USE AN960-416
MS21048-4 N

REQUIREMENTS PER ASSEMBLY/INSTALLATION -103	-101	PART OR IDENTIFYING NUMBER	
	32	MS1.54-10	S
	32	MS1.54-12	S
	44	MS21042-4	N
	42	AN960-416	W
-	1	-103	W
-	1	-101	W
1		-23	E
1	1	-21	B
	2	-19	G
	12	-17	CC
	1	-13	E
2		-13	C
1		-11	W
	28	-9	G
	2	-7	D
	2	-3	W
	-	-1	A

UNIT OF MEAS | ITEM OR FIND NO | CODE IDENT NO

UNLESS OTHERWISE SPECIFIED
ALL MACHINED SURFACES 125√

SEE D-70814 FOR ABBREVIATION CODE AND SYMBOL REQUIREMENTS

SEE D-70812 FOR STANDARD MANUFACTURING AND INSTALLATION REQUIREMENTS

ASSOCIATED LIST PROVIDED SEE DOCUMENT SAME NUMBER PREFIXED WITH FOR PARTS LIST

3 4 2

NOTES:

1. IMPRESSION MARKING OF PARTS NOT PERMITTED.
2. BREAK SHARP EDGES .03
3. MINIMUM CLEAN UP REQUIRED ON FAYING SURFACES OF ALL WELDED COMPONENTS
4. WELD PER MIL-W-8611
5. -9 GUSSETS TO BE EQUALLY SPACED (WITHIN ±.03) ALONG -17 COVER AS SHOWN
6. MAGNETIC INSPECT NO CRACKS ALLOWED
7. ACID RESIN PRIMER, USING PRIMER PER MIL-C-8514, APPLY PER 0-75149-1.
8. APPLY 0-75252-1 EPOXY PRIMER PER 0-75149-3
9. APPLY 0-75052-180 EPOXY ENAMEL PER 0-75149-4.
10. APPLY MIL-S-8802C-20 SEALANT TO FAYING SURFACES OF ALL BOLTED DETAILS.
11. USE AN960-416 WASHERS WITH NAS1134 SCREWS, NAS1154 SCREWS & MS21042-4 NUTS AS REQUIRED, TO INSURE INTERFACE CLAMPUP

Qty	Part or Identifying No.	Nomenclature or Description	Zone	Material / Notes
32	NAS1134-10	SCREW		
24	NAS1154-10	SCREW		
64	MS21042-4	NUT		
12	AN960-416	WASHER		SEE NOTE 11
1	-103	WELD ASSY		
1	-101	WELD ASSY		
1	-23	BOSS	B 4	M 1/2 x 1.90 DIA ASTM-A7 STEEL
1 1	-21	BOSS	B 4	M 1.22 x 1.50 DIA
2	-19	RAIL	E 0	M .27 x 3.21
2	-17	COVER	E 0	M .27 x 17.7 x 1.21
1	-13	END PLATE	B 0	M .27 x 17.7 x 3.32
2	-13	CLIP	B 3	M .27 x 2.4 x 1.76
1	-11	WEB	B 5	M .27 x 4.4 x 1.76
28	-9	GUSSET	C 0	M .27 x 1.5 x 1.5
2	-7	PLATE	C 0	M .27 x 1.5 x 3.8 ASTM-A7 STEEL
2	-3	WELD ASSY	C 0	
—	-1	ASSY		U NEXT ASSY 72C0160 (2 REQD)
72C0171-1				

Requirements per assembly installation · Unit of meas · Item or find no. · Code ident no. · Part or identifying number · Nomenclature or description · Zone · Note symbol · Stock size, material, specification, type (temper, etc.), heat treat

PARTS LIST

CONVAIR DIVISION OF GENERAL DYNAMICS
SAN DIEGO CALIFORNIA

END SUPT.- ASSY OF, BORON ALUMINUM WING BOX, TEST SPECIMEN

J 14170 72C0171

All machined surfaces 125 √

8

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REPORT DOCUMENTATION PAGE		READ INSTRUCTIONS BEFORE COMPLETING FORM
1. REPORT NUMBER CASD-NSC-73-005	2. GOVT ACCESSION NO.	3. RECIPIENT'S CATALOG NUMBER
4. TITLE (and Subtitle) Design and Fabrication of a Boron/Aluminum Composite Wing Box Test Specimen		5. TYPE OF REPORT & PERIOD COVERED Final, March 1972-May 1973
		6. PERFORMING ORG. REPORT NUMBER
7. AUTHOR(s) William F. Wennhold		8. CONTRACT OR GRANT NUMBER(s) N62269-72C-0414
9. PERFORMING ORGANIZATION NAME AND ADDRESS Convair Aerospace Division, General Dynamics Corp, San Diego Operation		10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS Task Number WF41422206 Work Unit Number HJ202
11. CONTROLLING OFFICE NAME AND ADDRESS Aero Structural Department 18974 Naval Air Development Center, Warminster, PA		12. REPORT DATE July, 1973
		13. NUMBER OF PAGES 113
14. MONITORING AGENCY NAME & ADDRESS(if different from Controlling Office)		15. SECURITY CLASS. (of this report) Unclassified
		15a. DECLASSIFICATION/DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Approved for public release. Distribution unlimited.

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

Boron/Aluminum, Composite, Metal Matrix, Wing Box

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

A wing box test specimen utilizing boron/aluminum as the principal structural material was designed, fabricated and successfully tested.

The primary purpose of the program was to demonstrate the feasibility of utilizing boron/aluminum in a major aircraft structural element. The wing section was typical of many aircraft wing structures and simulated an integral wing fuel tank section capable of withstanding internal pressure in addition to the basic bending and

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torsion loads.

A detailed stress analysis was performed. Computer programs were utilized to generate internal member loads and web shears, and to analyze load introduction areas. Conventional aircraft stress analysis techniques were utilized throughout.

The wing box test specimen was fabricated using the latest metal matrix composite fabrication techniques which, for the most part, are based upon conventional sheet metal technology.

Structural testing was accomplished at the Naval Air Development Center, Warminster, Pennsylvania, where the specimen successfully withstood 150% of limit loads in bending, torsion, combined bending and torsion, and internal pressure. It was then subjected to a Navy fighter fatigue spectrum for over five 6,000 hour life times.

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